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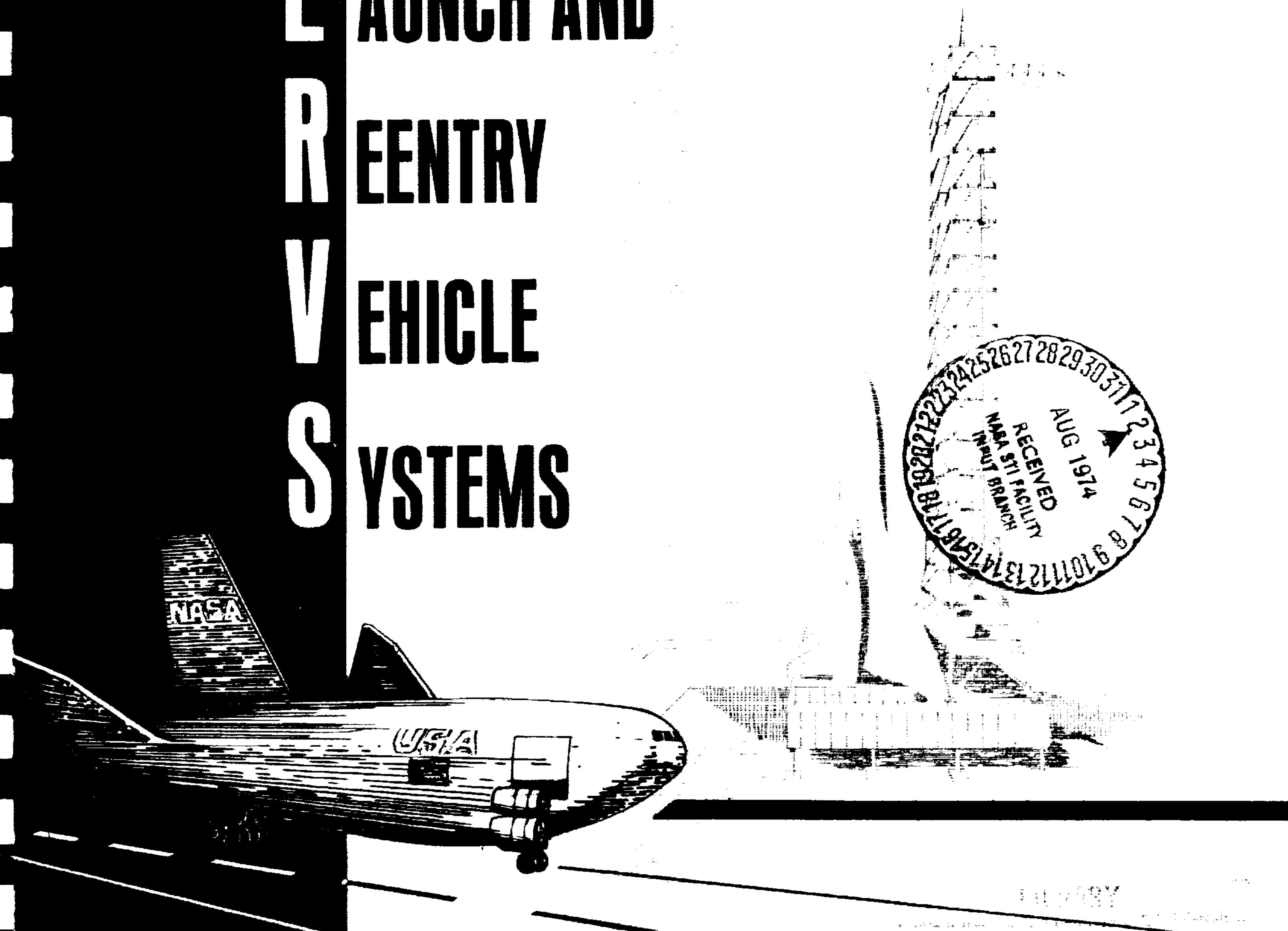
REPORT MDC E0049
CONTRACT NAS 9-9204
NOVEMBER 1969

INTEGRAL LAUNCH AND REENTRY VEHICLE SYSTEMS



(NASA-CR-66866) INTEGRAL LAUNCH AND REENTRY VEHICLE SYSTEMS. VOLUME 4: ONE AND A HALF STAGE CONFIGURATION Final Report (McDonnell-Douglas Astronautics Co.) 140 p

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INTEGRAL LAUNCH AND REENTRY VEHICLE SYSTEM

NASA-CR-66866
REPORT MDC E0049
CONTRACT NAS 9-9204
NOVEMBER 1969
SERIAL NO. 169

VOLUME IV

ONE AND A HALF STAGE CONFIGURATION

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MCDONNELL DOUGLAS
CORPORATION

FOREWORD

This volume of McDonnell Douglas Astronautics Company Report Number MDC E0049 constitutes a portion of the final report for the "Integral Launch and Reentry Vehicle Systems Study". The study was conducted by the MDAC for the NASA-Langley Research Center under Contract NAS9-9204.

The final report consists of the following:

Executive Summary

Vol. I - Design, Configuration and Subsystems

Vol. II - Performance, Aerodynamics, Mission and Operations

Vol. III - Plans, Costs, Schedules, Technologies

Vol. IV - One and a Half Stage

McDonnell Douglas Astronautics Company gratefully acknowledges the cooperation of the companies which provided technical assistance during this study. They are:

Pratt & Whitney Aircraft Division, United Aircraft Corporation

Rocketdyne Division, North American Rockwell Corporation

This study was managed and supervised by:

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of McDonnell Douglas Astronautics Company.

ABSTRACT

This study emphasized a two stage to orbit reusable spacecraft system for use in transporting cargo and passengers to and from a near earth orbital space station. A single conceptual "point" design was treated in detail and several alternate systems, corresponding to alternate payloads (size and weight), were examined based on parametric excursions from the "point" design. The overall design goal was to configure the carrier and orbiter vehicles to minimize operational and program recurring costs. This goal was achieved through high system reliability, vehicle recoverability,and rapid ground turnaround capability made possible through modular replaceable component design and use of an integrated onboard self test and check-out system. Launch and land landing of both stages at the ETR launch site was a study groundrule as was the nominal 25,000 lb payload delivered to and returned from orbit and packaged in a 15 ft. diameter by 30 ft. long cylindrical canister. The resulting system has a gross lift-off weight of 3.4 million pounds.

The Orbiter is a 107 ft. HL-10 configuration, modified slightly in the base area to accommodate the two boost engines. The launch propellant tanks are integral with the primary body structure to maximize volume available for propellant.

The Carrier is a 195 ft. clipped delta configuration with ten launch engines identical to those of the orbiter. A dual lobed cylindrical launch propellant tank forms the primary body structure. A 15% thick delta wing is incorporated which contains the landing gear, airbreathing engines and propellant.

A broad range of weight, cost and performance sensitivity data were generated for the baseline and alternate system designs. Pertinent development and resource requirements were identified, development and operational schedules were prepared and corresponding recurring and non-recurring cost data were estimated. Program plans were outlined for the design, manufacture and testing of the Orbiter and Carrier vehicles and for the pursuit of critical technologies pacing vehicle development.

Stage and a half and reusable systems employing expendable launch vehicles were considered initially, but, these efforts were subsequently terminated prior to completion. The expendable launch vehicle data are reported separately. The stage and a half effort employed a version of the McDonnell Douglas Model 176 with four drop tanks.

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1.0 INTRODUCTION

The Stage-and-one-half was one of the initial space vehicle concepts considered in this study. Work was begun on the design of the concept to the initial study groundrules. The special study requests from the NASA to alter the design to satisfy various groundrule changes, pertaining primarily to payload size and geometry, resulted in several configurations. Thus, a "baseline" configuration was never established as such. By direction from the NASA, all effort on the stage-and-one-half design was terminated in August 1969, and emphasis was shifted to the two-stage fully-reusable concept. Accordingly, a considerable amount of stage-and-one-half design data, primarily parametric in nature, were generated to various sets of groundrules, without arriving at a recommended configuration or any specific design conclusion. This volume is a compilation of these stage-and-one-half design data generated up to the point of study termination. No attempt has been made to add to or further integrate these data. The data of this volume are, perforce, imcomplete and should therefore not be used for comparison with other concepts. This discussion is broken into section as follows:

The conceptual design section contains the results of the identification and definition of candidate concepts, sizing analyses for various payloads and studies directed toward optimizing vehicle performance and configuration description.

Performance analyses includes the aerodynamic characteristics for three different length 176M vehicles; trajectory analyses and performance for powered ascent phase and unpowered reentry and glide phase; studies of the sensitivity of payload to sizing parameters and inert weight uncertainties; and the effect of impulsive velocity and orbit inclination on payload capability.

The operational section contains the mission profile and sequence of events for the baseline vehicles. Results of investigations of various operational modes such as the swing-nose concept, vehicle-payload integration, and alternate mission are reported.

Results of the preliminary parametric cost analysis for the 1-1/2-stage concept are reported in the last section of this volume. The analyses include a summary of program cost estimates, parametric studies of total program recurring cost, and cost sensitivities. All costs are gross preliminary figures because the design, development and operational programs were not defined to sufficient depth in the 1-1/2-stage study. The preliminary cost analysis performed for this study is reported in MDAC Report H367, "Integral Launch and Reentry Vehicle System - Final Report", Volume I, 29 July 1969.

2.0 CONCEPTUAL DESIGN DEFINITION

The conceptual design definition effort for the 1-1/2 stage vehicle concept is presented in this section. The effort consisted of candidate concept definition, sizing for various payloads, and studies directed toward optimizing vehicle performance. The design analysis was terminated prior to the completion of the study by direction of the NASA. This discussion, therefore, includes only a report of the effort that was completed and no final baseline description is presented. The basic concept employs a "core" vehicle (i.e. the basic spacecraft) for orbital operations and reentry, plus expendable tip tanks which provide propellant for boost in addition to that contained in the core vehicle. A cross feed system is employed to transfer the propellant from the tip tanks to the engine system in the core vehicle.

2.1 Definition of Candidate Concepts - The preliminary reentry vehicle (R/V) shapes that were considered for this study are shown in Figures 2-1 through 2-3. Previous MDAC studies have shown the basic 176M shape to be a very efficient approach to providing good hypersonic performance, while at the same time maintaining a relatively high volumetric and packaging efficiency. The configuration in Figure 2-1 emphasizes high volumetric utilization at a nominal hypersonic L/P (1.4) while the configuration in Figure 2-2 provides increased hypersonic performance (L/D = 2.4) with some compromise in efficiency alumetric. The geometric properties of these shapes were determined for comparison purposes and are shown in Figures 2-4 through 2-11. The wetted areas and volumes are derived by integrating the area under the curves defined by P/L vs. X/L and A/L^2 vs. X/L, respectively. The lengths, areas and volumes are defined non-dimensionally (X/L, A/L^2, V/L^3) to provide a rapid means of converting to true values for any given vehicle size. The moments for volume and wetted area were similarly determined by integrating the areas under the curves on either side of the reference (zero-moment) station. The area between the volume curve and the reference c.g. line is equal in both sides of that line, for example, in Figure 2-10.

The baseline core vehicle shape that was selected for further study and optimization is shown in Figure 2-12. The rationale for the selection is discussed in the following sections. The basic geometric curves are the same as Figures 2-8 through 2-11. This configuration is designated the 176M-ILRV.

176 M-1.4 CONFIGURATION

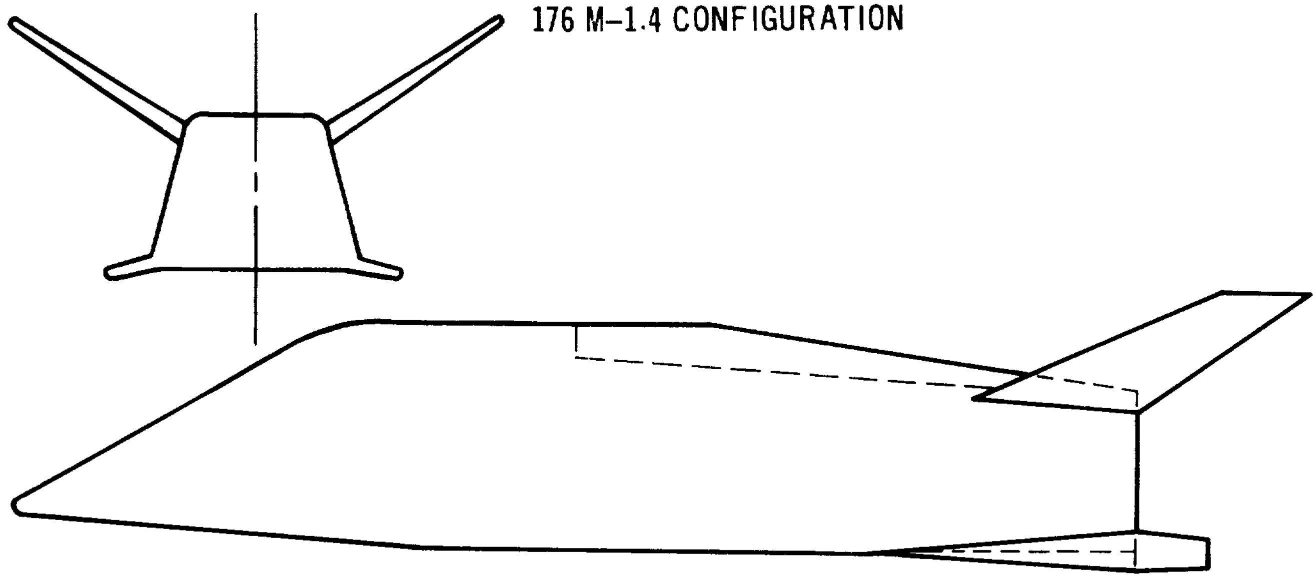


Figure 2-1

176 M-2.4 CONFIGURATION

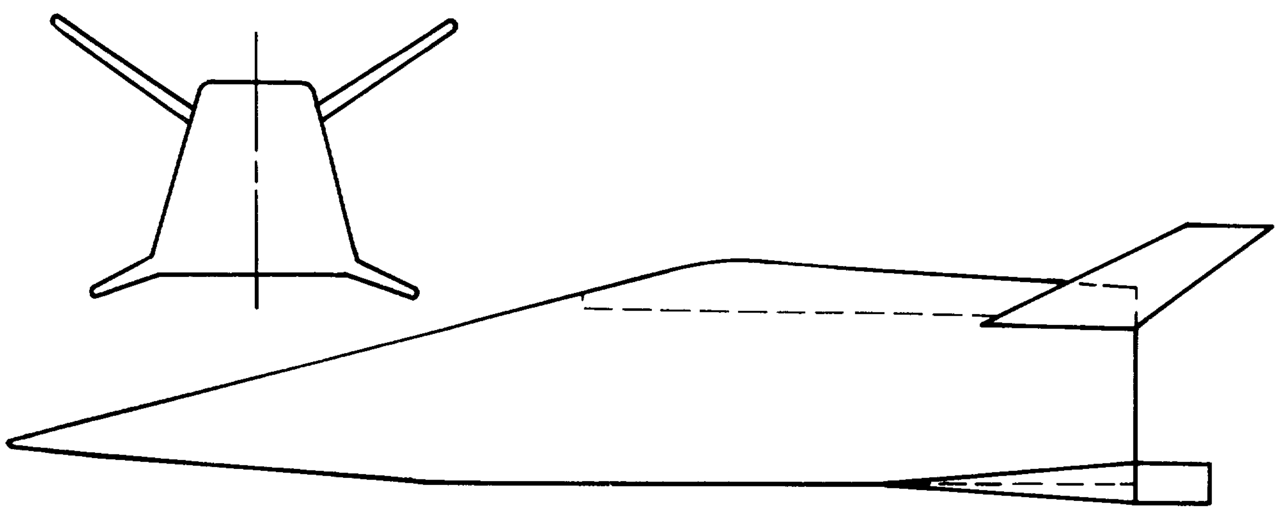


Figure 2-2

176 M - ILRV CONFIGURATION

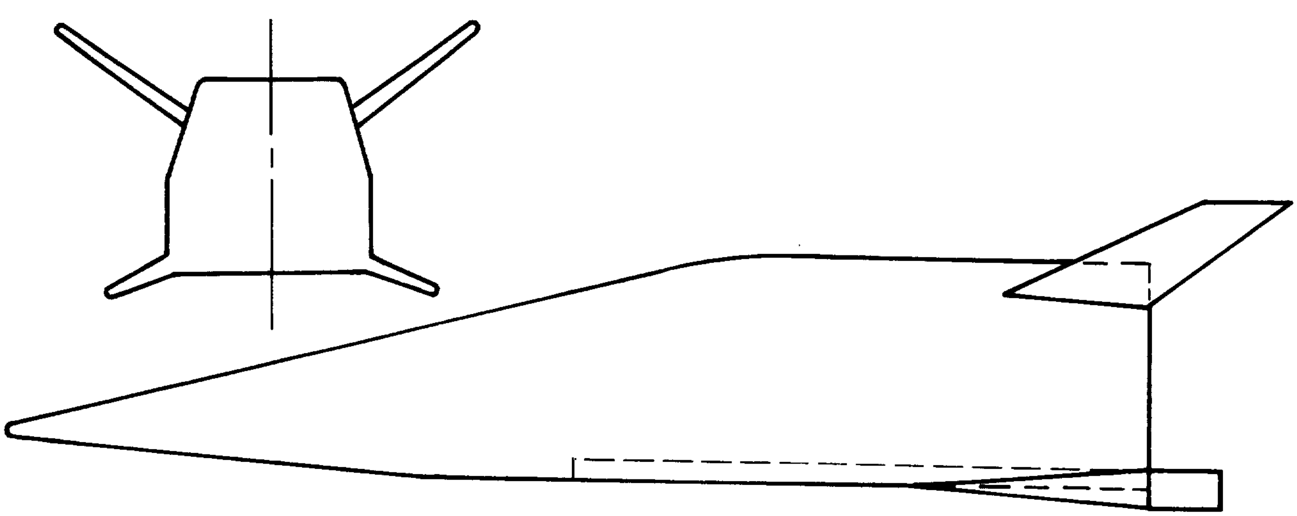


Figure 2-3

WETTED AREA – 176M-1.4 CONFIGURATION
(Outer ML – Body Only)

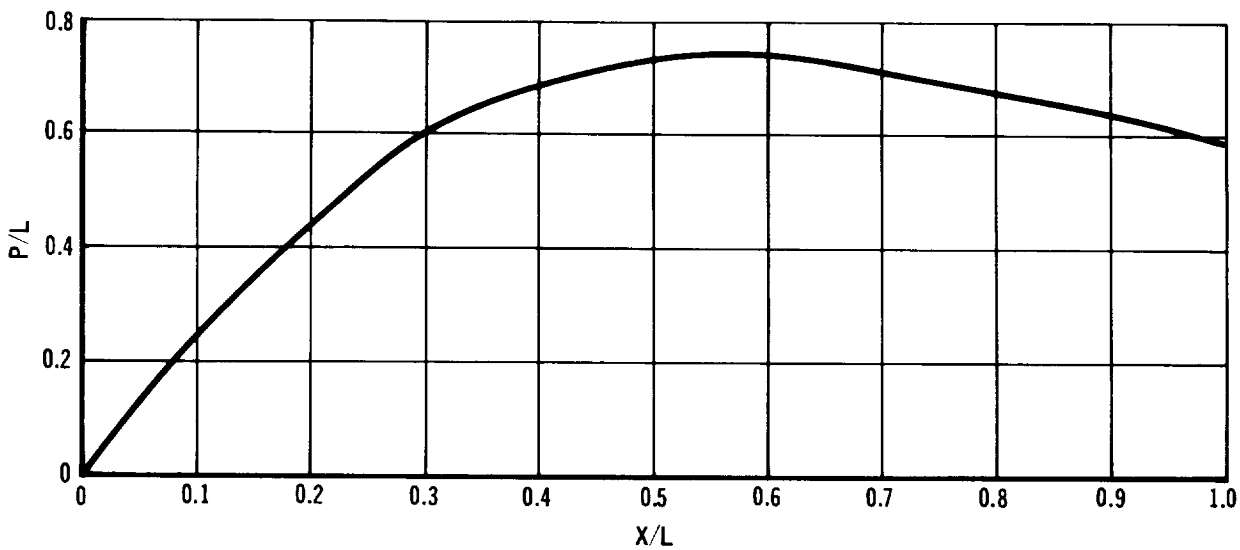


Figure 2–4

VOLUME - 176M-1.4 CONFIGURATION
(Outer ML - Body Only)

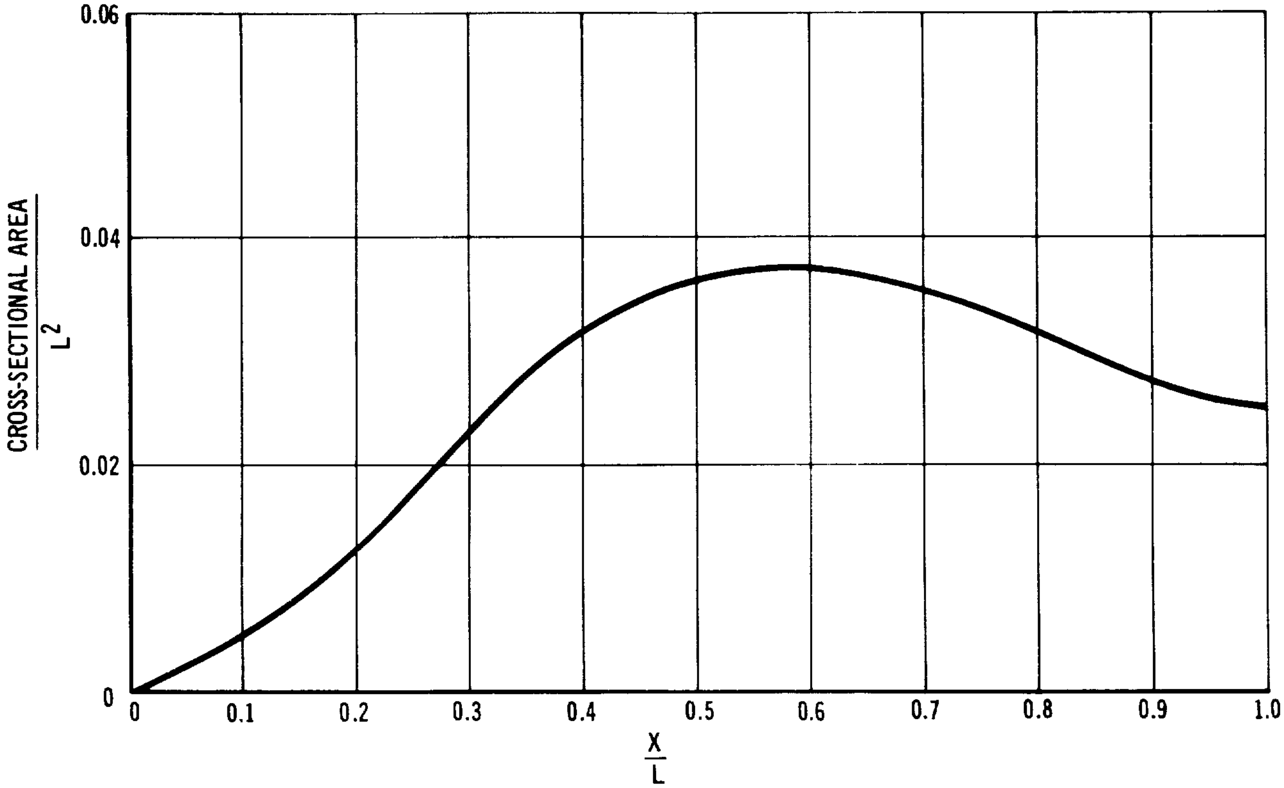


Figure 2-5

WETTED AREA – 176 M–2.4 CONFIGURATION
(Body Only)

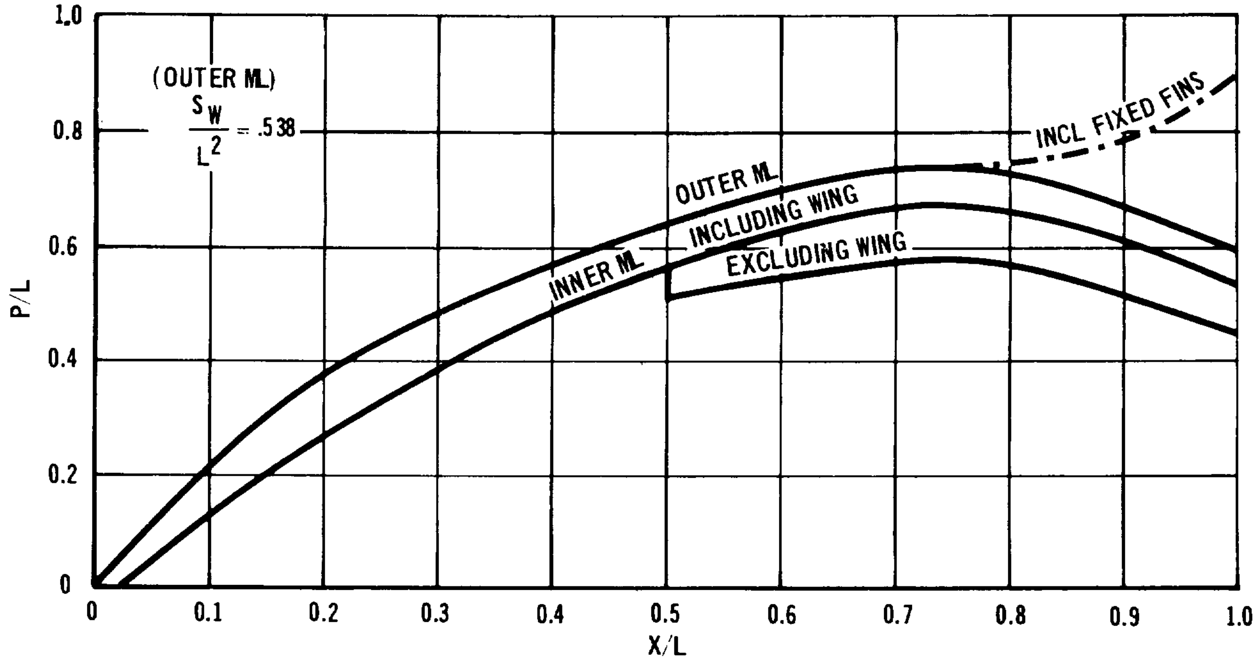


Figure 2–6

VOLUME – 176 M-2.4 CONFIGURATION
(Body Only)

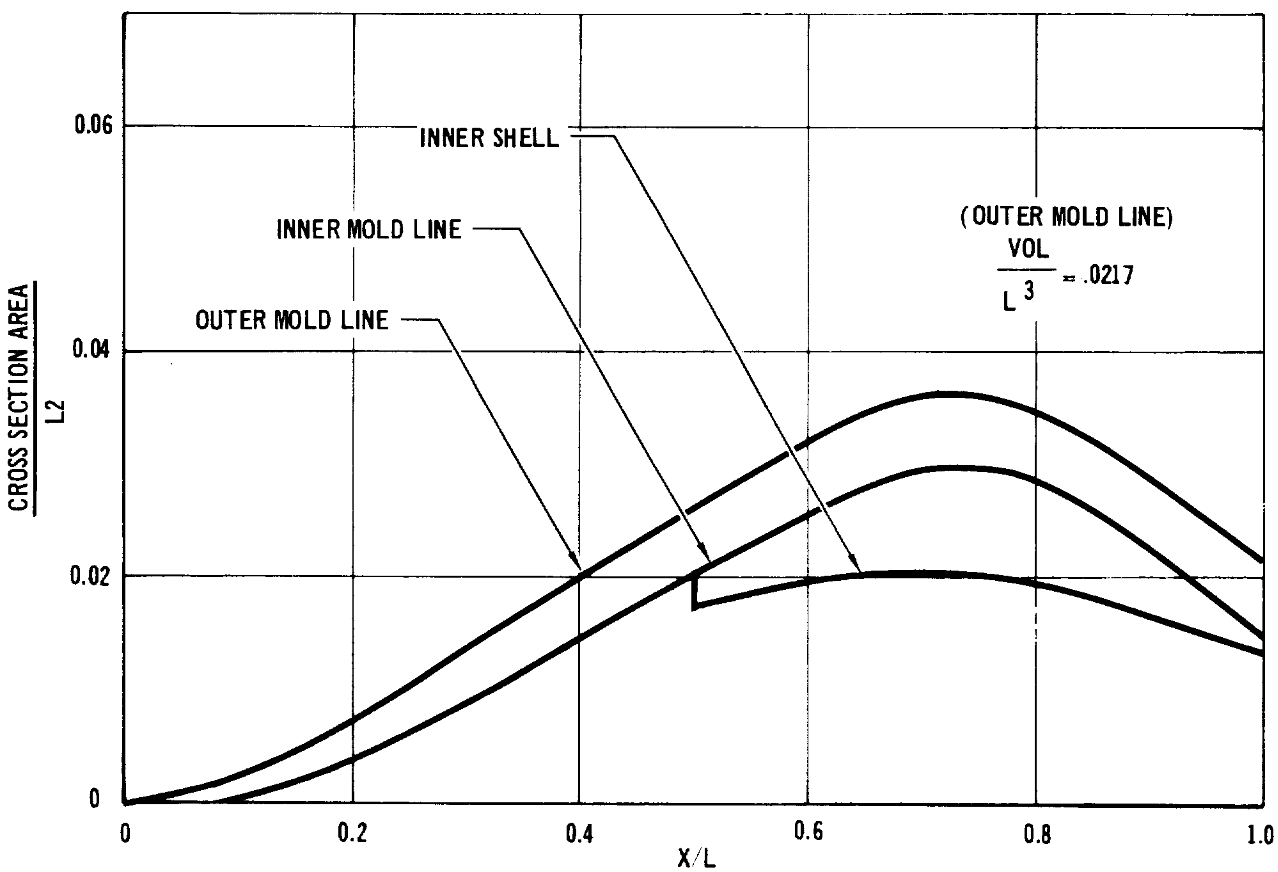


Figure 2-7

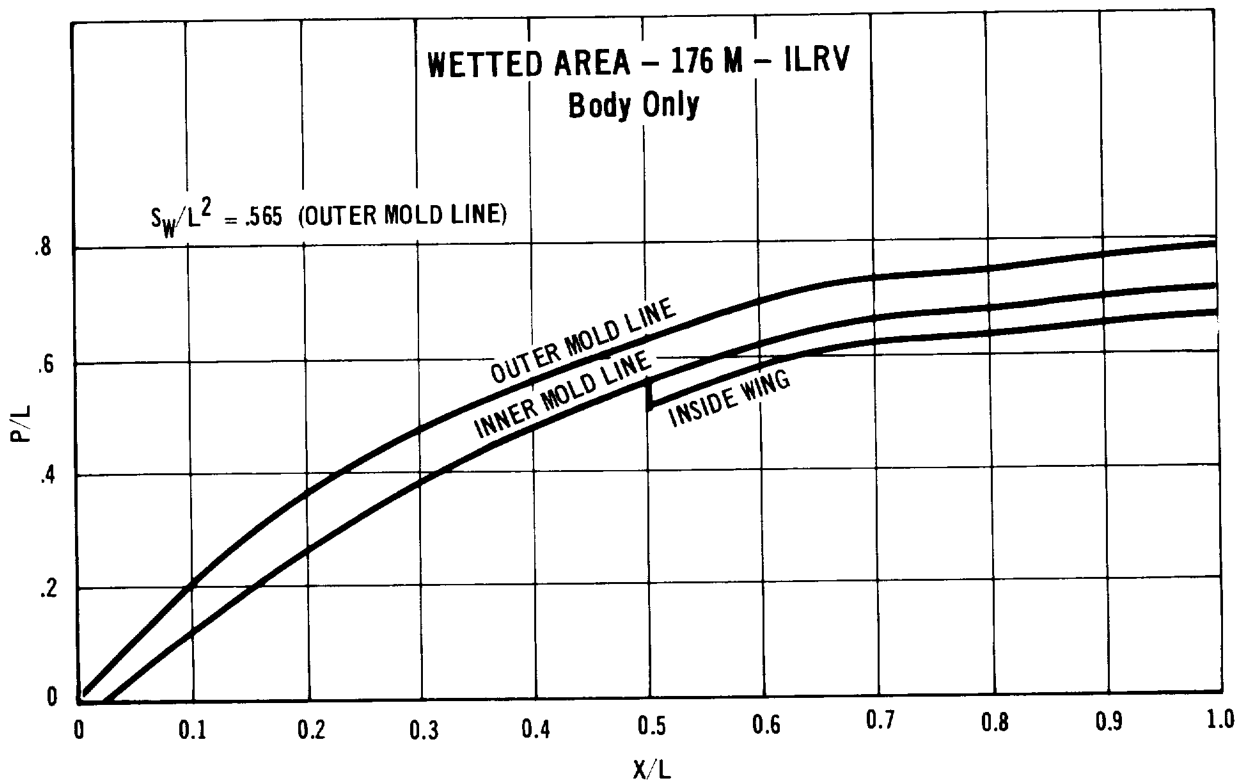
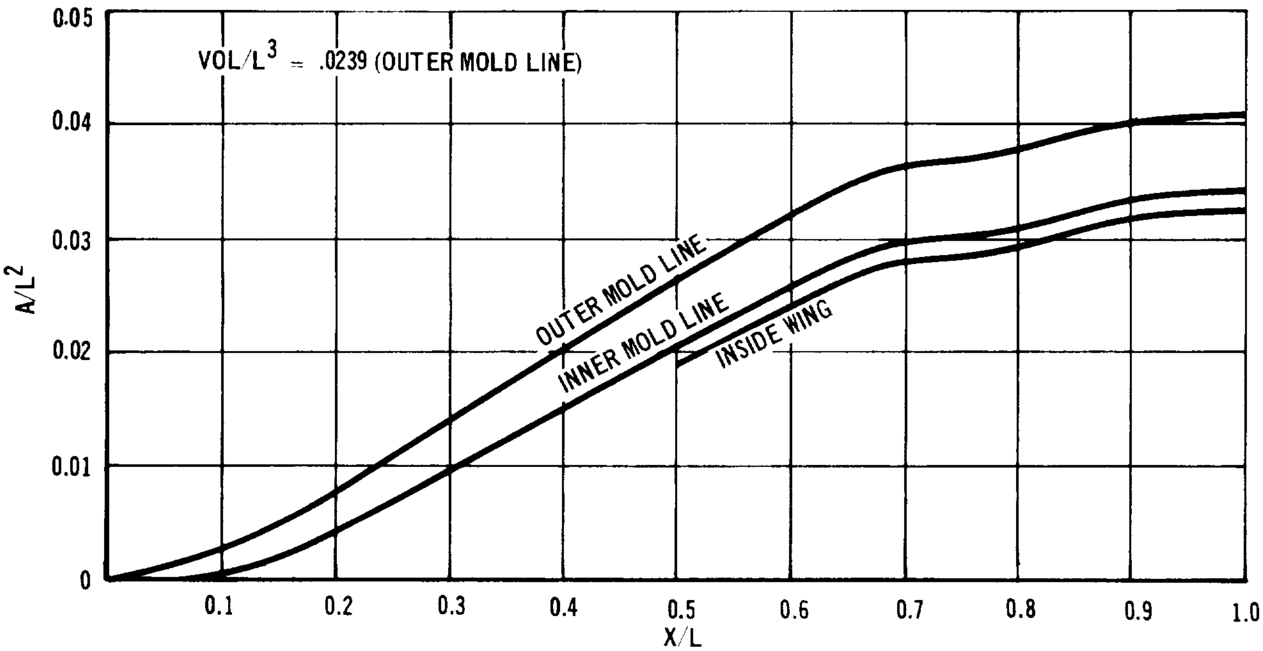

WETTED AREA – 176 M – ILRV
Body Only
S_W/L^2 = .565 (OUTER MOLD LINE)
OUTER MOLD LINE
INNER MOLD LINE
INSIDE WING
P/L
X/L
0
.2
.4
.6
.8
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0

Figure 2-8

VOLUME – 176 M – ILRV
Body Only



TV -

Figure 2–9

WETTED AREA MOMENT
176 M – ILRV
Outer ML – Body Only

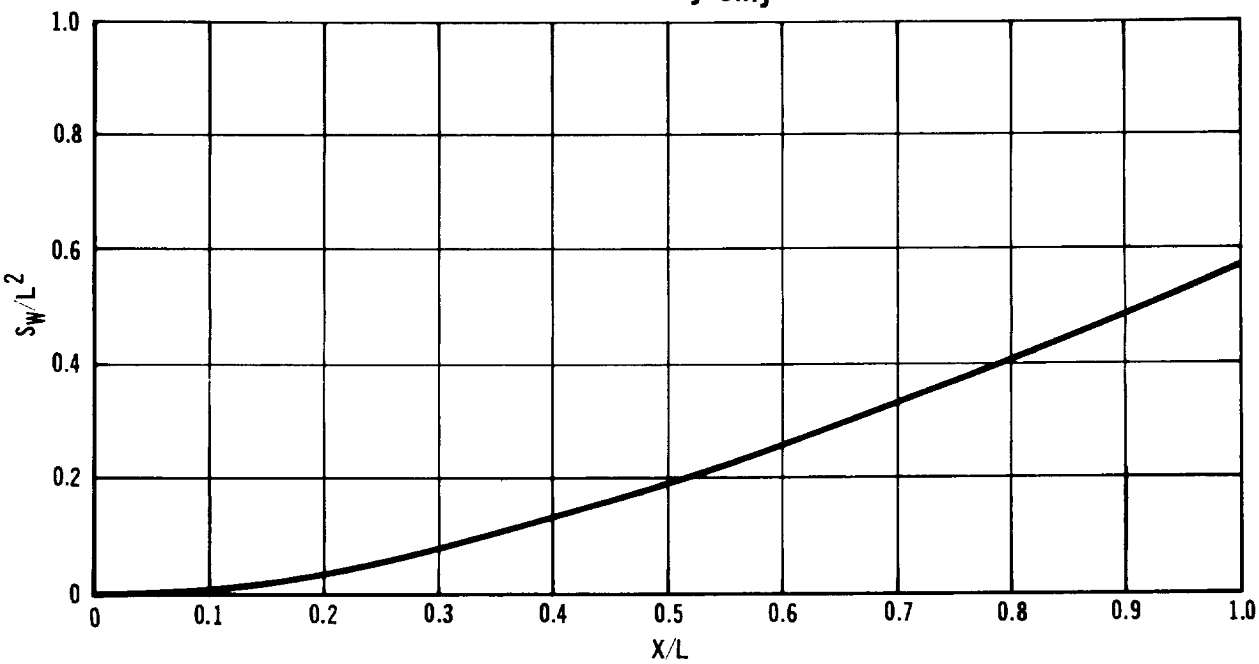


Figure 2-10

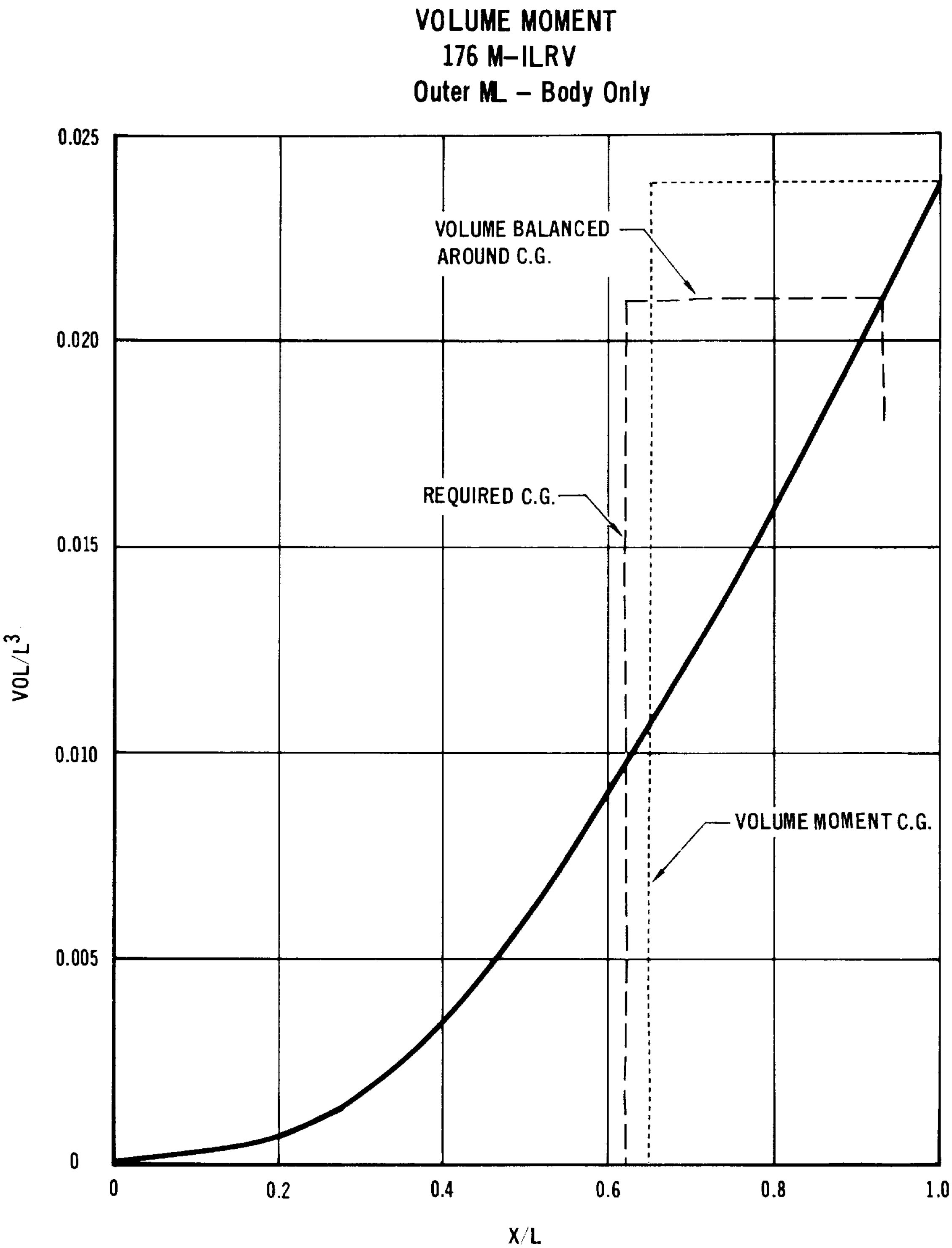

VOLUME MOMENT
176 M-ILRV
Outer ML – Body Only
0.025
0.020
0.015
0.010
0.005
0
VOL/L^3
0
0.2
0.4
0.6
0.8
1.0
X/L
VOLUME BALANCED AROUND C.G.
REQUIRED C.G.
VOLUME MOMENT C.G.

Figure 2-11

176 M – ILRV BASELINE

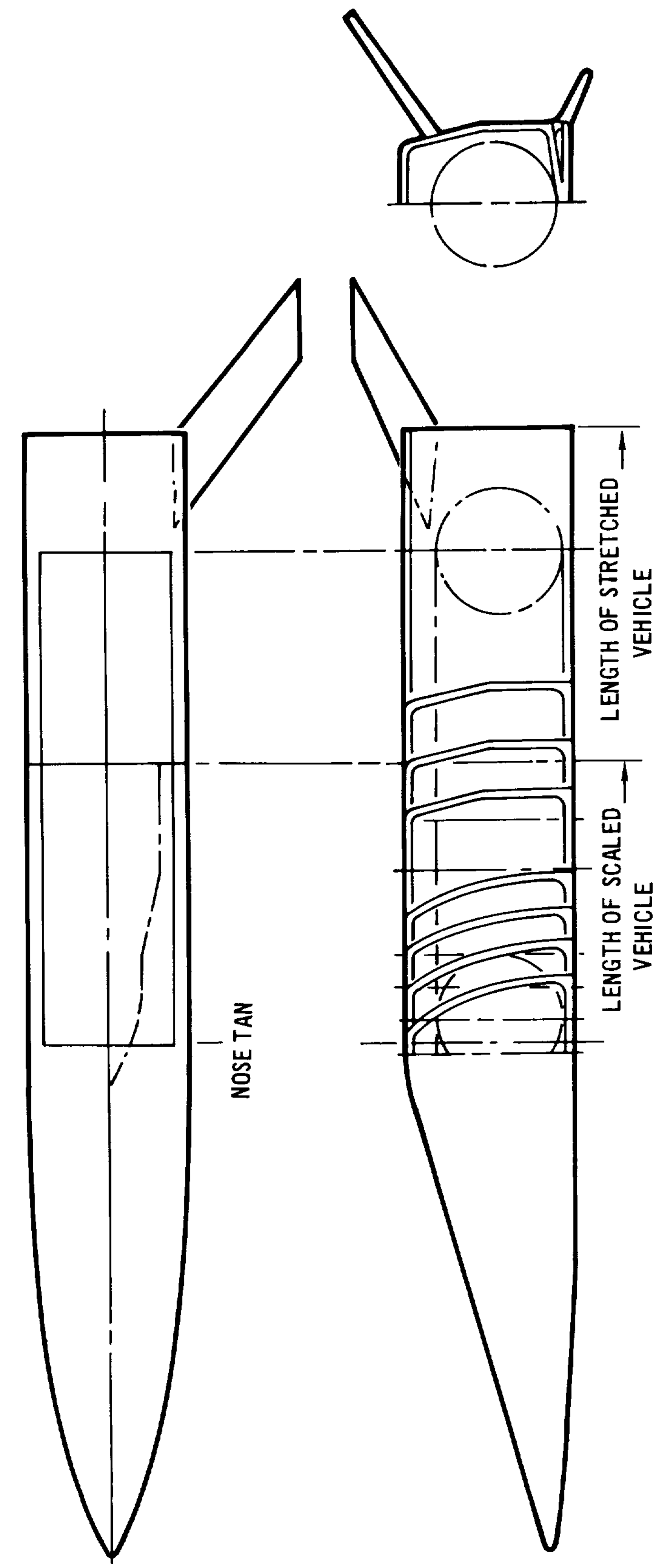


Figure 2–12

2.2 Vehicle Sizing - Preliminary studies were oriented towards sizing the core vehicle for various payload configurations. The core vehicle mold line surface, at approximately 75% of the length, is parallel to the spacecraft centerline around the entire periphery of the body. This permits stretching the spacecraft length without increasing body cross-sectional characteristics. Payload configurations with geometric constraints may then be accommodated by scaling the vehicle to obtain a required cross-section and then stretching to attain the desired length. This permits flexibility in design to attain a reentry vehicle which will satisfy a wide variation of constraints while minimizing the unuseable spacecraft volume and maintaining or improving aerodynamic performance. To further improve volumetric utilization and facilitiate the installation of an integral boost engine system, maximum vehicle base area is required which motivated the modification shown in Figure 2-3.

Figures 2-13 through 2-16 show the internal arrangement, for four payload conditions, with locations for cargo, crew, and an engine compartment. The geometrically constrained cargos include an installation clearance. Basic geometry characteristics are tabulated and overall dimensions shown. The spacecraft lengths are derived combining scaling and stretching.

Figures 2-17 through 2-20 show the launch configuration for the corresponding cargo characteristics specified. A tabular weight summary is shown, along with overall dimensions of spacecraft and tanks. (NOTE: The weights presented in the volume were derived using a MDAC computerized weights model. Further these weights include ablative heat protection for all vehicles and the thermal insulation is sized for maximum cross range trajectories.) For each configuration, the two sets of tanks have equal dimensions for manufacturing commonality considerations. The first set of tanks (side-mounted) provides approximately the same characteristic velocity as the second set (top-and-bottom-mounted). The bottom tank is sized to permit 90° rotation of the hinged nose section. The advantages of this capability are for payload delivery, future mission requirements, pre-launch operations, etc.

Figure 2-21 illustrates the inboard profile or general arrangement of the 130 ft. (1,560 in.) core vehicle. This vehicle accommodates a crew of up to 12 and a 15.0 ft. diameter by 60.0 ft. long cargo container. The figures shows the installation of the heavier equipment as far forward as possible to establish a balanced vehicle. The nose landing gear is also installed as far forward as design permits. The main landing gear is retracted into the area immediately aft

ILRV–81.5 FT CORE VEHICLE
Cargo: 25,000 Lbs

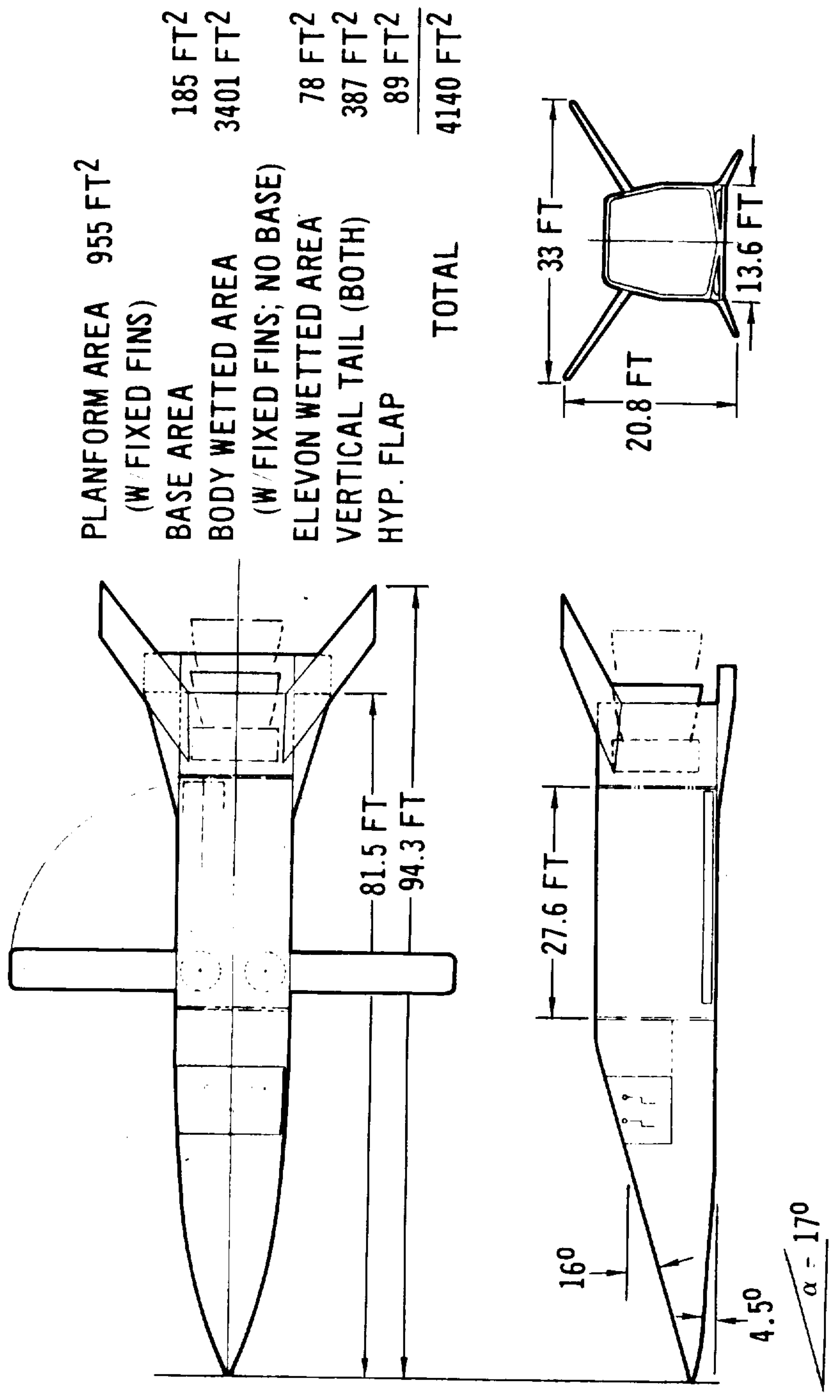


Figure 2–13

ILRV–95.3 FT CORE VEHICLE
CARGO: 25,000 LB

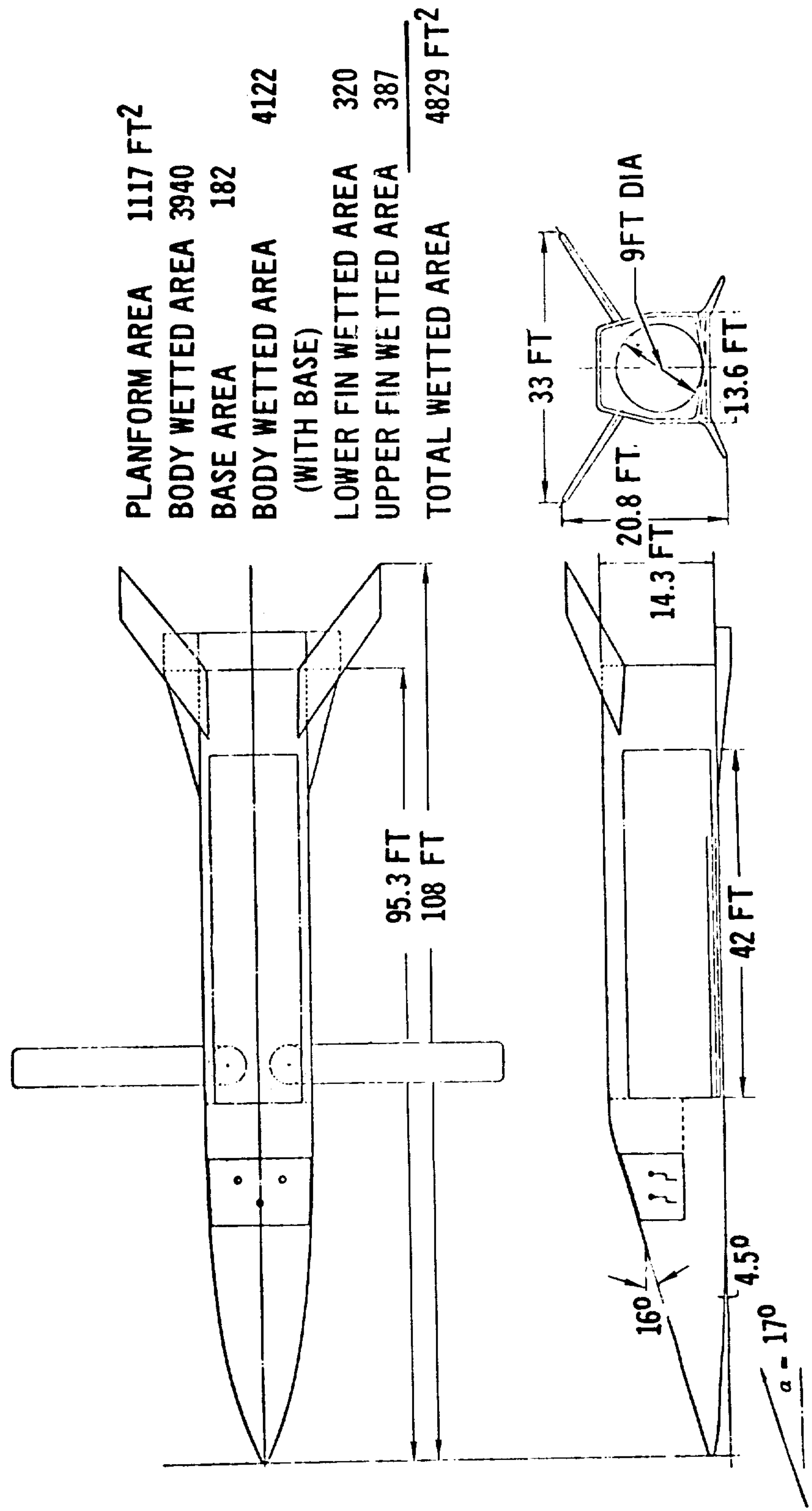


Figure 2–14

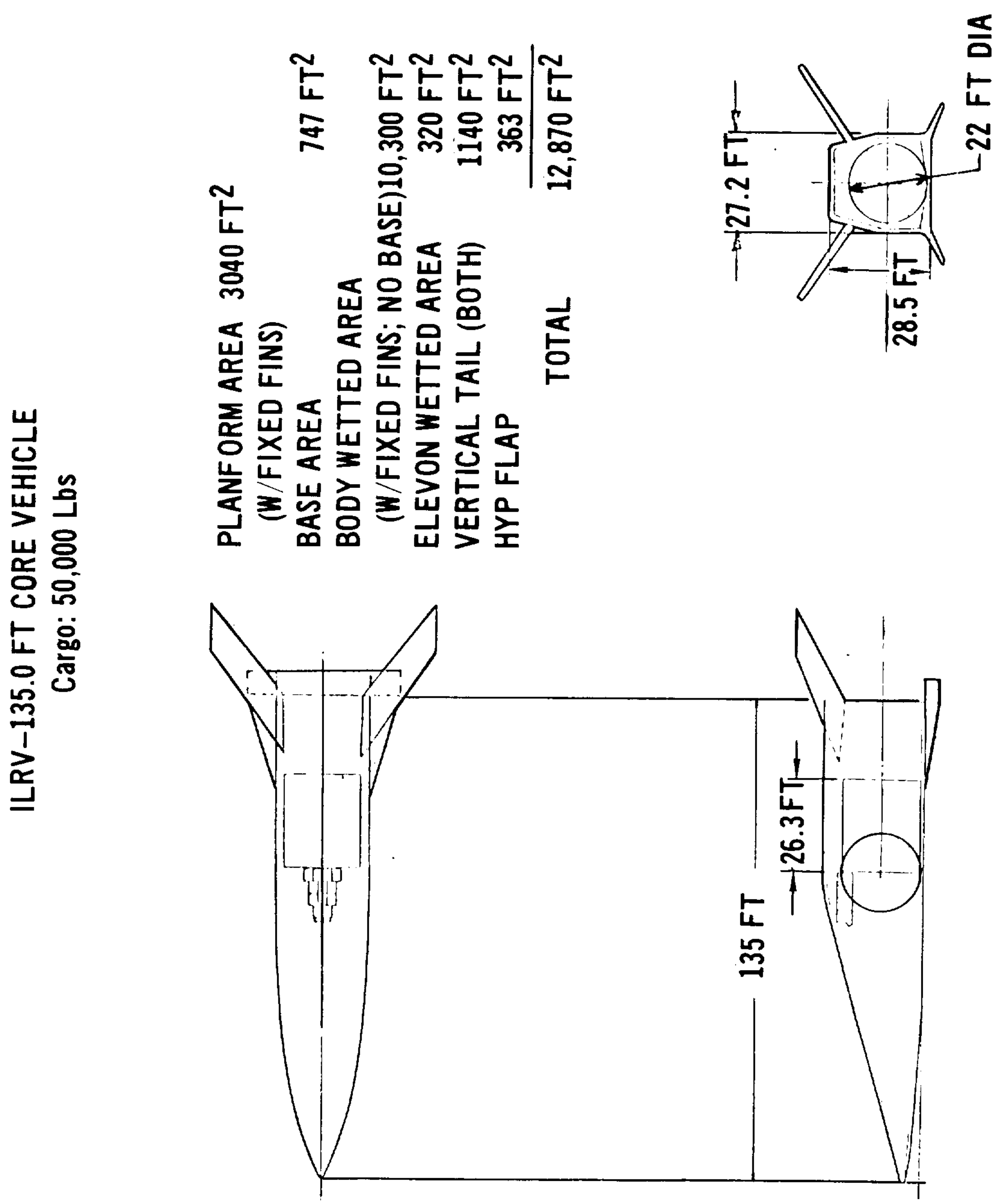

ILRV–135.0 FT CORE VEHICLE
Cargo: 50,000 Lbs
PLANFORM AREA 3040 FT²
(W/FIXED FINS)
BASE AREA 747 FT²
BODY WETTED AREA
(W/FIXED FINS; NO BASE) 10,300 FT²
ELEVON WETTED AREA 320 FT²
VERTICAL TAIL (BOTH) 1140 FT²
HYP FLAP 363 FT²
TOTAL 12,870 FT²
27.2 FT
28.5 FT
22 FT DIA
26.3 FT
135 FT

Figure 2–15

ILRV–130.0 FT CORE VEHICLE
Cargo: 50,000 LBS

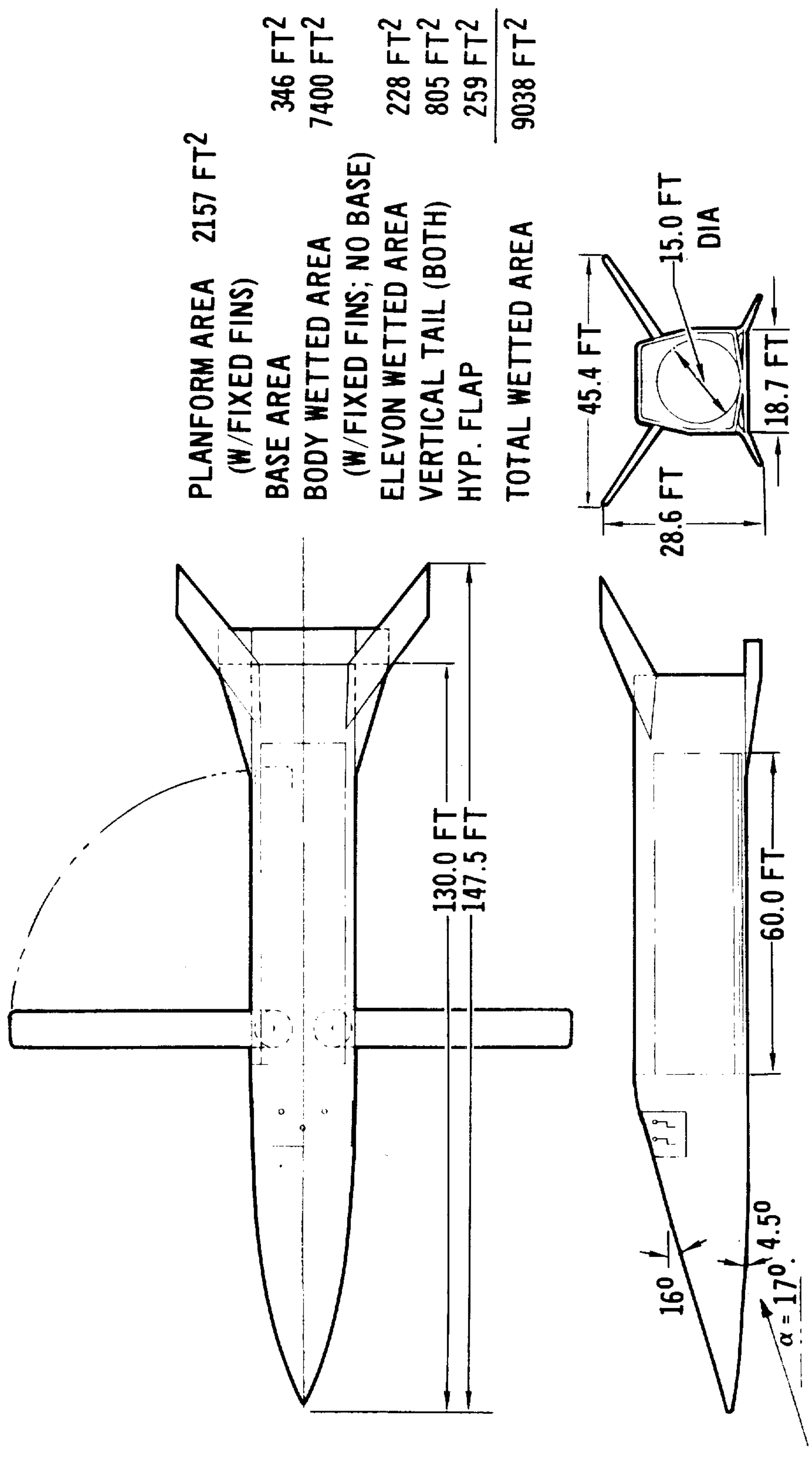


Figure 2–16

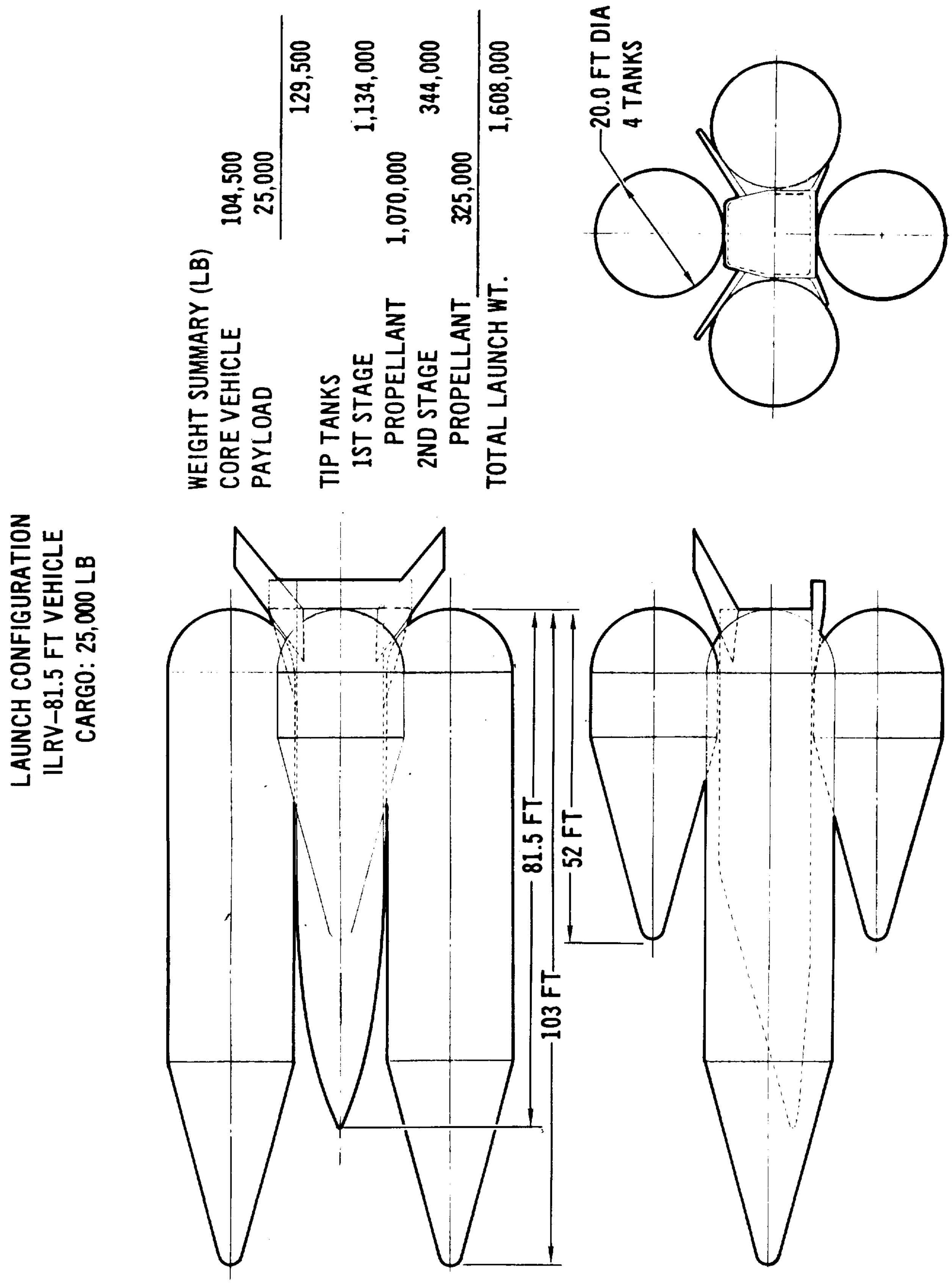

LAUNCH CONFIGURATION
ILRV–81.5 FT VEHICLE
CARGO: 25,000 LB
WEIGHT SUMMARY (LB)
CORE VEHICLE 104,500
PAYLOAD 25,000
129,500
TIP TANKS
1ST STAGE
PROPELLANT 1,070,000 1,134,000
2ND STAGE
PROPELLANT 325,000 344,000
TOTAL LAUNCH WT. 1,608,000
20.0 FT DIA
4 TANKS
81.5 FT
52 FT
103 FT

Figure 2–17

LAUNCH CONFIGURATION
ILRV–95.3 FT VEHICLE

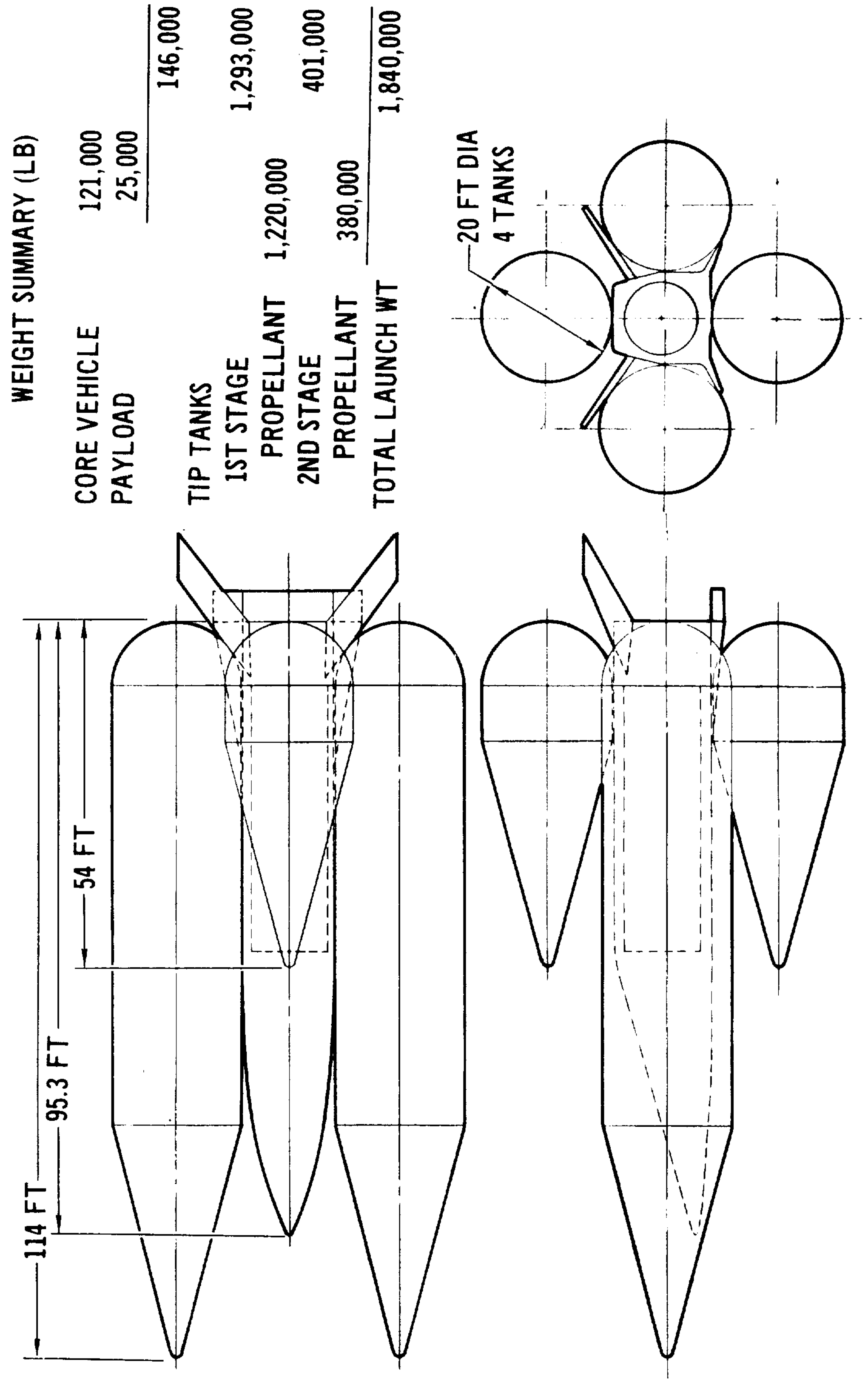


Figure 2–18

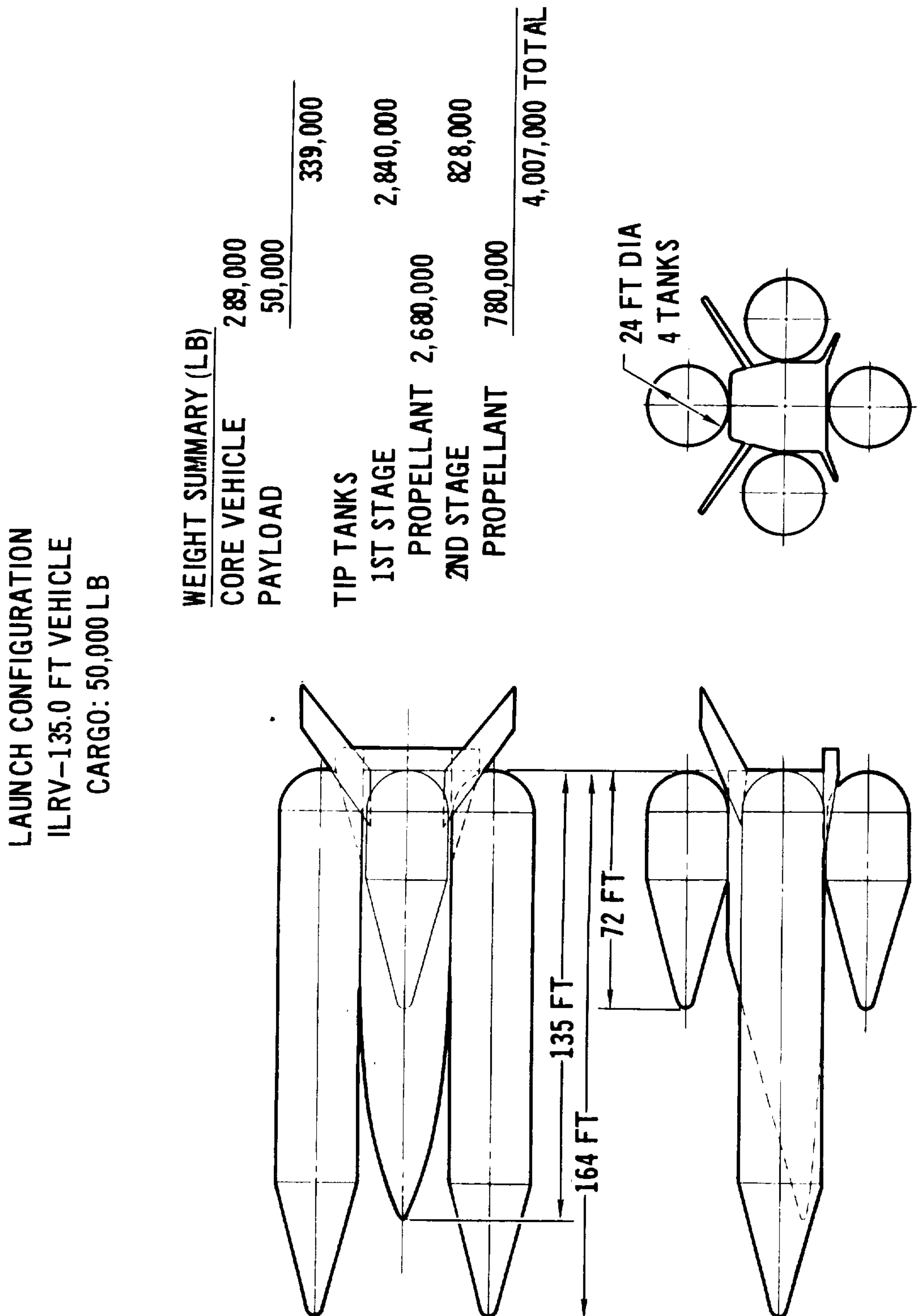
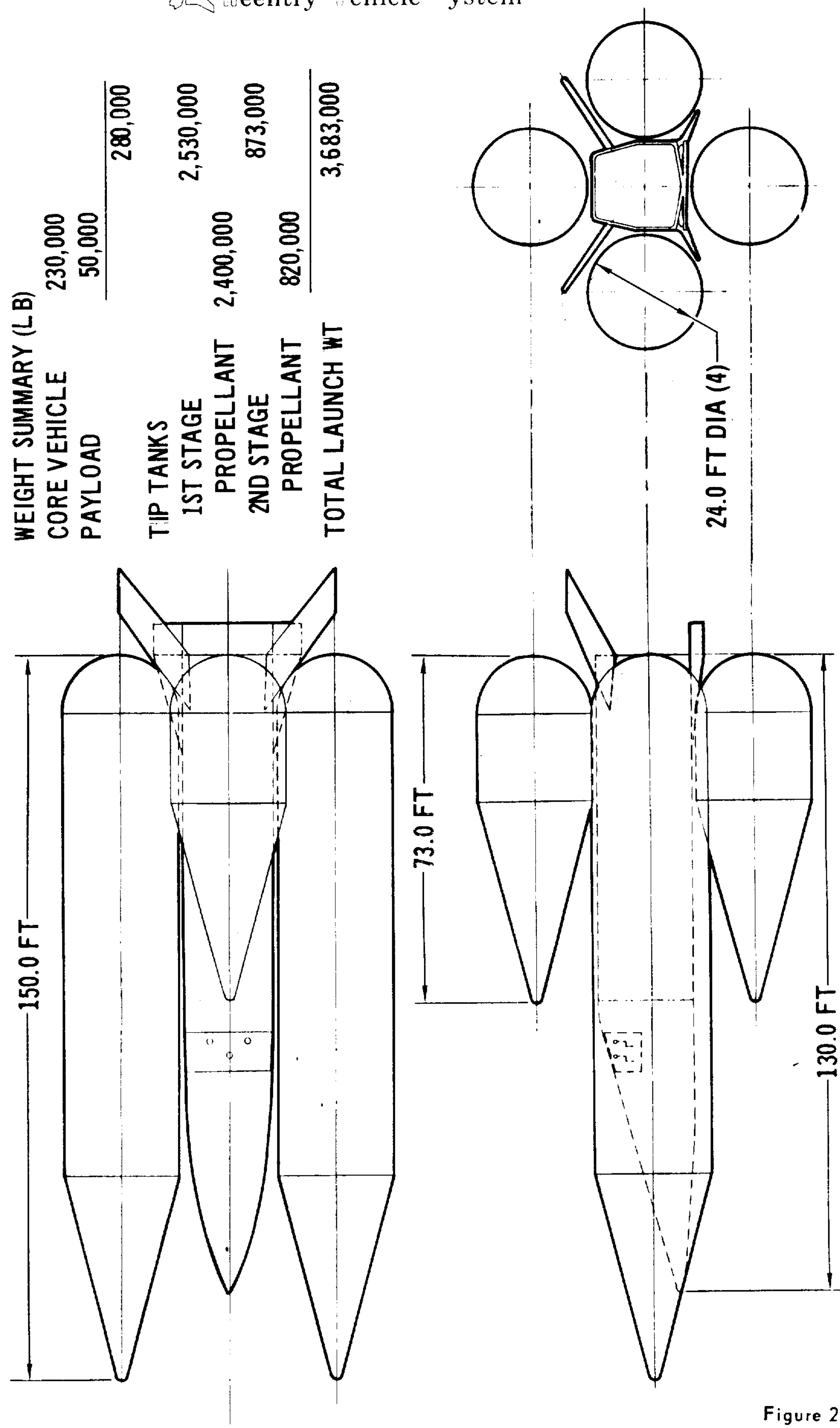

LAUNCH CONFIGURATION
ILRV-135.0 FT VEHICLE
CARGO: 50,000 LB
WEIGHT SUMMARY (LB)
CORE VEHICLE 289,000
PAYLOAD 50,000
339,000
TIP TANKS
1ST STAGE
PROPELLANT 2,680,000
2,840,000
2ND STAGE
PROPELLANT 780,000
828,000
4,007,000 TOTAL
24 FT DIA
4 TANKS
72 FT
135 FT
164 FT

Figure 2-19

LAUNCH CONFIGURATION
ILRV-130.0 FT VEHICLE
CARGO: 50,000 LB



WEIGHT SUMMARY (LB)		
CORE VEHICLE	230,000	
PAYLOAD	50,000	
		280,000
TIP TANKS		
1ST STAGE PROPELLANT	2,400,000	2,530,000
2ND STAGE PROPELLANT	820,000	873,000
TOTAL LAUNCH WT		3,683,000

Figure 2-20

ILRV–130 FT (1560 IN.) VEHICLE
Inboard Profile

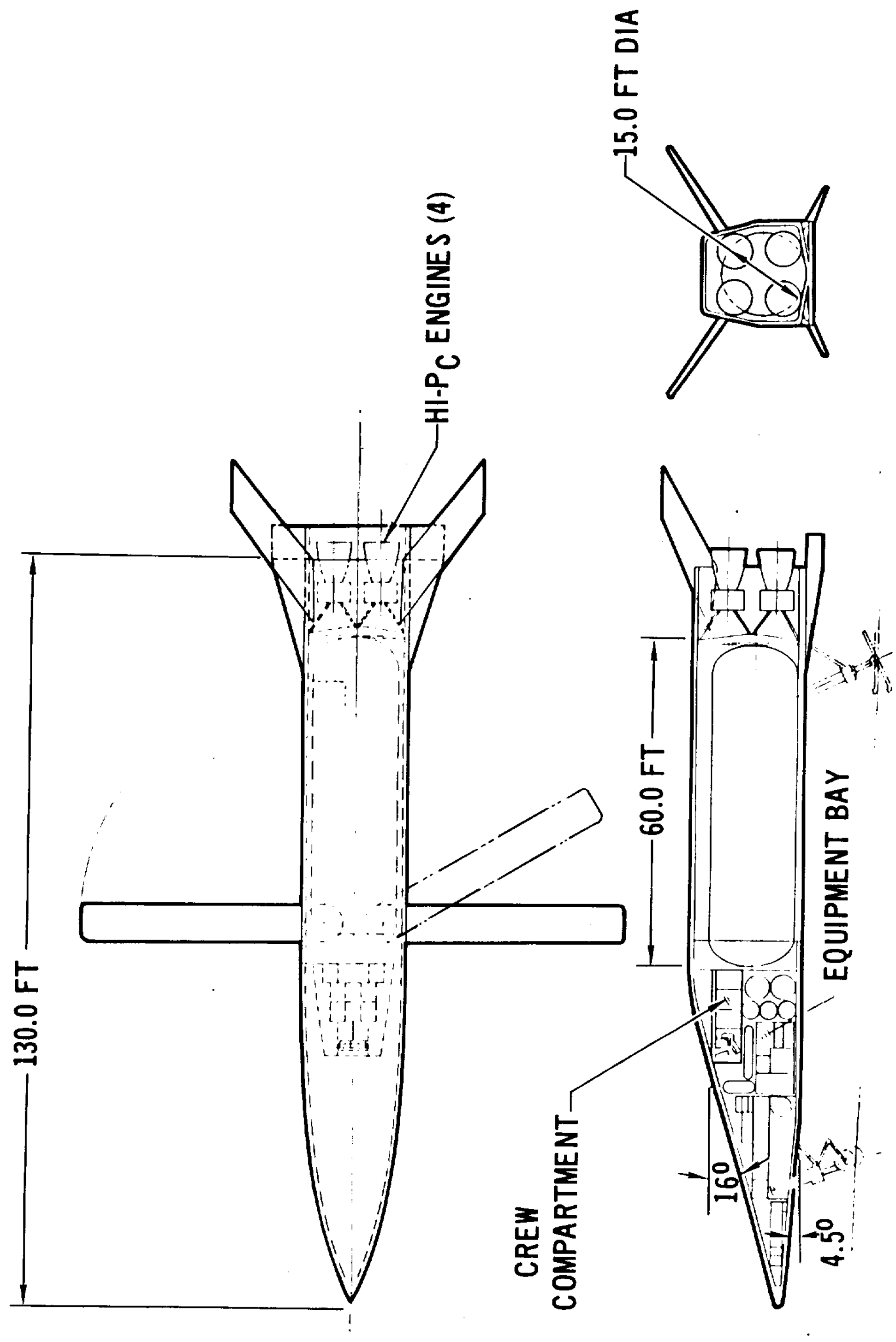


Figure 2–21

of the cargo container and variable geometry wing and forward of the four (4) main Hi Pc engines.

2.3 Design For Reduced Plan Loading - The vehicles shown in Section 2.2 had high entry plan loadings (> 100 psf) and high landing velocities (> 200 knots), due to the loaded density of the vehicle. The configuration was analyzed with the objective of reducing the entry plan loadings to a level compatible with entry temperatures less than 2200°. This requires entry loadings in the vicinity of 60 to 65 psf. The basic methods employed were generally a combination of scaling up the vehicle, enlarging the lower fixed fin area, and employing structural weight savings associated with the lower temperature environments resulting from the reduced plan area loadings. To provide for systematic configuration modifications, a model was defined as shown in Figure 2-22. This model possesses a simple pyramidal nose and rectangular cross-section after body.

For the first type of modifications, four specific spans were investigated and obtained by extending the nose leading edge (a straight line element). For the span, b_1, the model resembles the 176M configuration shown in Figure 2-16. As the span is stretched the plan shape approaches a delta.

The overall size of the model was selected to provide a minimum envelope for a 15 ft. by 60 ft. cylinder weighting 50,000 lb. The same technique may also be applied to a vehicle sized for 25,000 lb. payload in a 15 ft. diameter, 30 ft. long container. As the span increased, the abse area increased and the span increments were chosen to provide a ratio of base width, b, to base height, h, of 1, 1.5, 2 and 3.

A second type of modification was examined consisting of scaling up the vehicles whose base width to height ratios were 1, and R. Scaled body lengths varied from 125 ft. to 180 ft.

The data of Table 2-1 describes the specific conditions for analysis for both the span stretching and the scaling modifications. As the model was stretched in a spanwise direction or was scaled up in length, the additional internal volume that was generated was filled with propellant tankage at a volumetric efficiency of 70 percent and limited only in front by the crew cabin aft bulkhead station and in the rear by the front of the engine bay.

GEOMETRY FOR SPAN STRETCHING AND SCALING

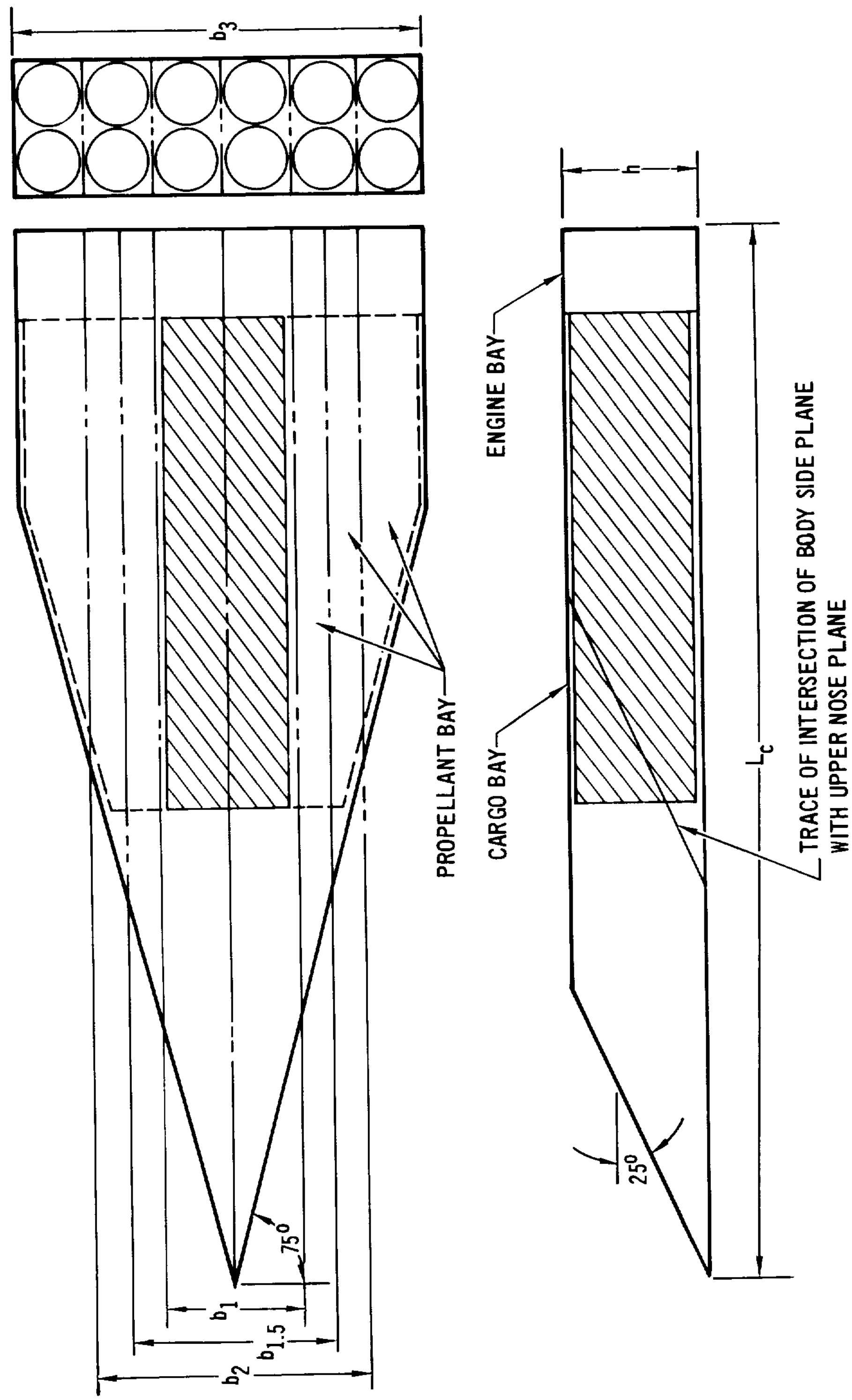


Figure 2-22

TABLE 2-1

CONDITIONS FOR ANALYSIS

SPAN STRETCHING

CASE	b/h	L_c
A-1	1.0	125
B-1	1.5	125
C-1	2.0	125
D-1	3.0	125

SCALING

CASE	b/h	L_c
A-1	1.0	125
A-2	1.0	140
A-3	1.0	180
C-1	2.0	125
C-2	2.0	140
C-3	2.0	180

The internal impulsive velocity of the spacecraft was determined and the remainder, required to provide a total of 31,000 fps, was assigned to four external tip tanks. Liquid oxygen/hydrogen propellants were assumed. Total external tank volumes, propellant loadings, gross lift-off weights and thrust required were determined. Propulsion system weights were estimated and the effect of this system size was iterated through the vehicle gross weight and thrust required analysis.

The gross weight at lift-off and the core vehicle weight at entry is shown in Figure 2-23 as a function of base area (span stretching). The gross weight increases significantly in going from a 1 x 1 base (280 sq. ft.) to a 3 x 1 base (840 sq. ft.). In moving toward the base proportions of approximately 4 x 1 (S_B = 1120 sq. ft.), the planform approaches a pure delta and volume increases approach zero. Wetted area and the structural weight goes up at a faster rate than the volume for propellants. The core vehicle weight at entry (minus boost phase propellants) also increases but at a decreasing rate for reasons described above.

The gross weight at lift-off and the core vehicle weight at entry as they are affected by scaling are shown in Figure 2-24 for the 1 x 1 vehicles and the 2 x 1 vehicles. As with span stretching, the weights increase with scaling.

The plan loadings at the atmospheric entry condition varies with the base area (span strethcing) as indicated in Figure 2-25. The top solid curve is based on 100% return cargo and the dashed middle curve is for no return cargo. The bottom dashed line is for the case of removing the entire propulsion system with 100 percent return cargo. The strong influence of the boost propulsion system size on

GROSS LIFTOFF AND CORE VEHICLE WEIGHT TRENDS

$\Lambda = 75^0$

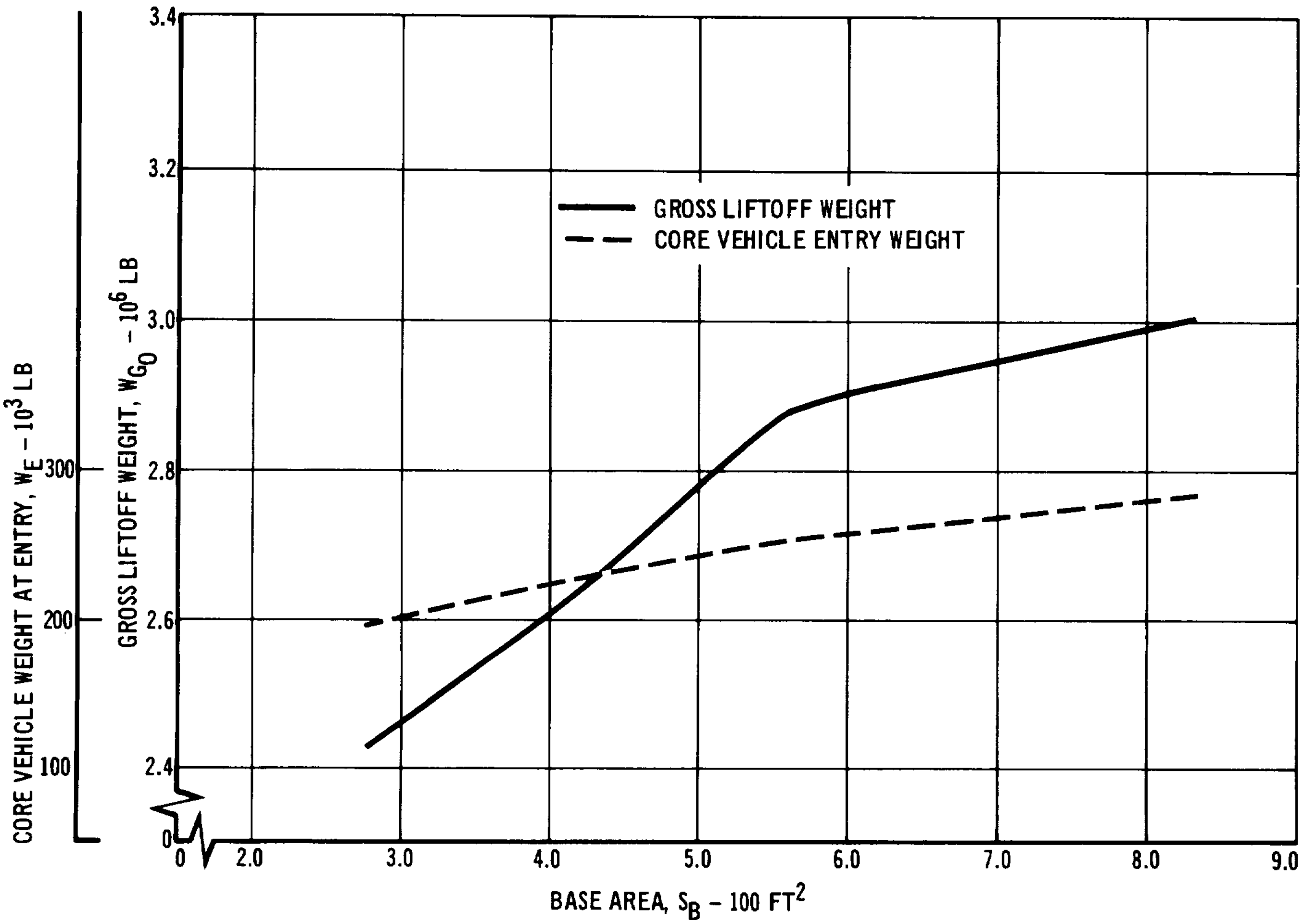


Figure 2-23

VEHICLE WEIGHT TRENDS

$\Lambda = 75^0$

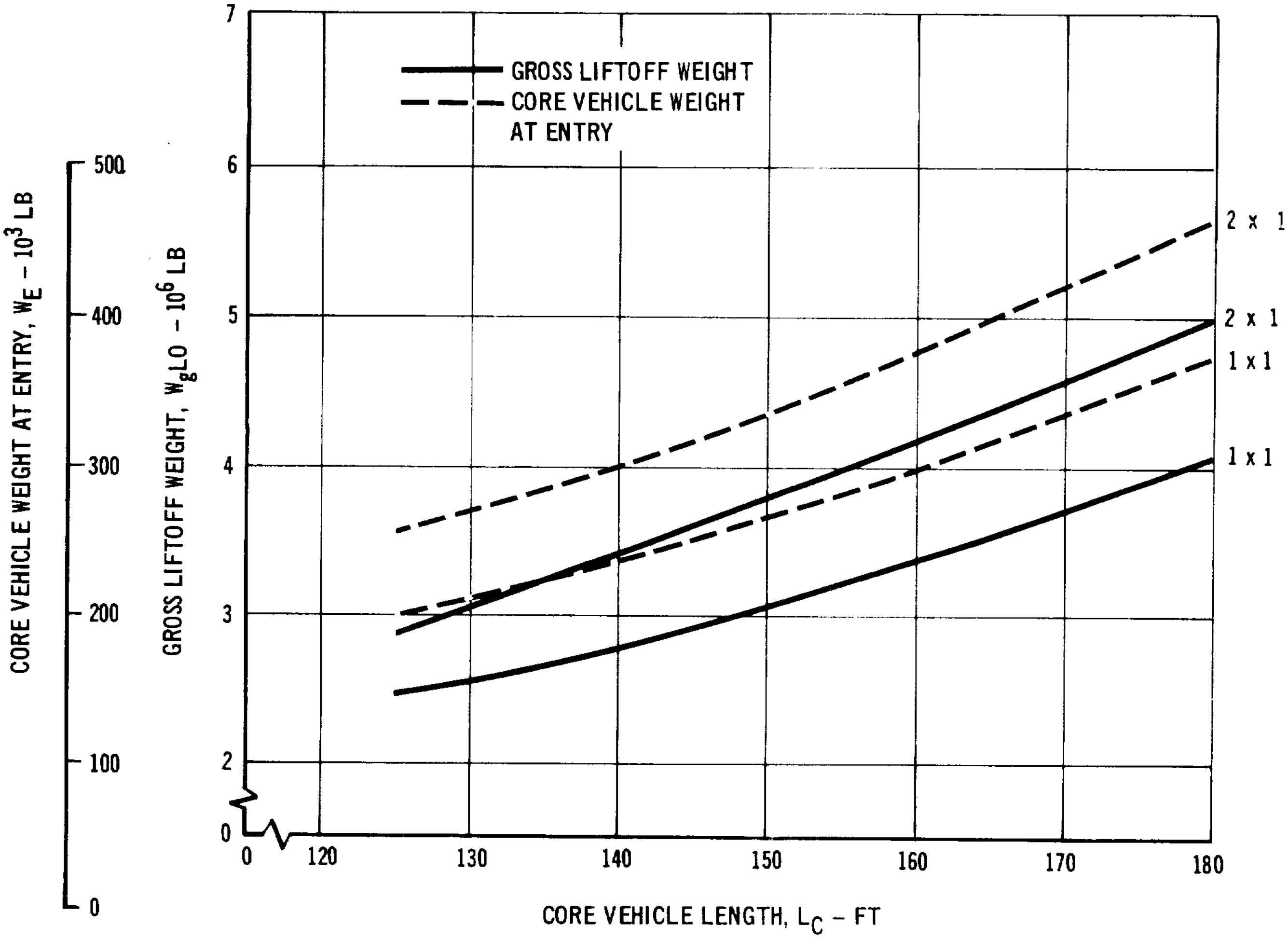


Figure 2-24

EFFECT OF SPAN STRETCHING
L = 125 Ft
$\Lambda = 75^0$

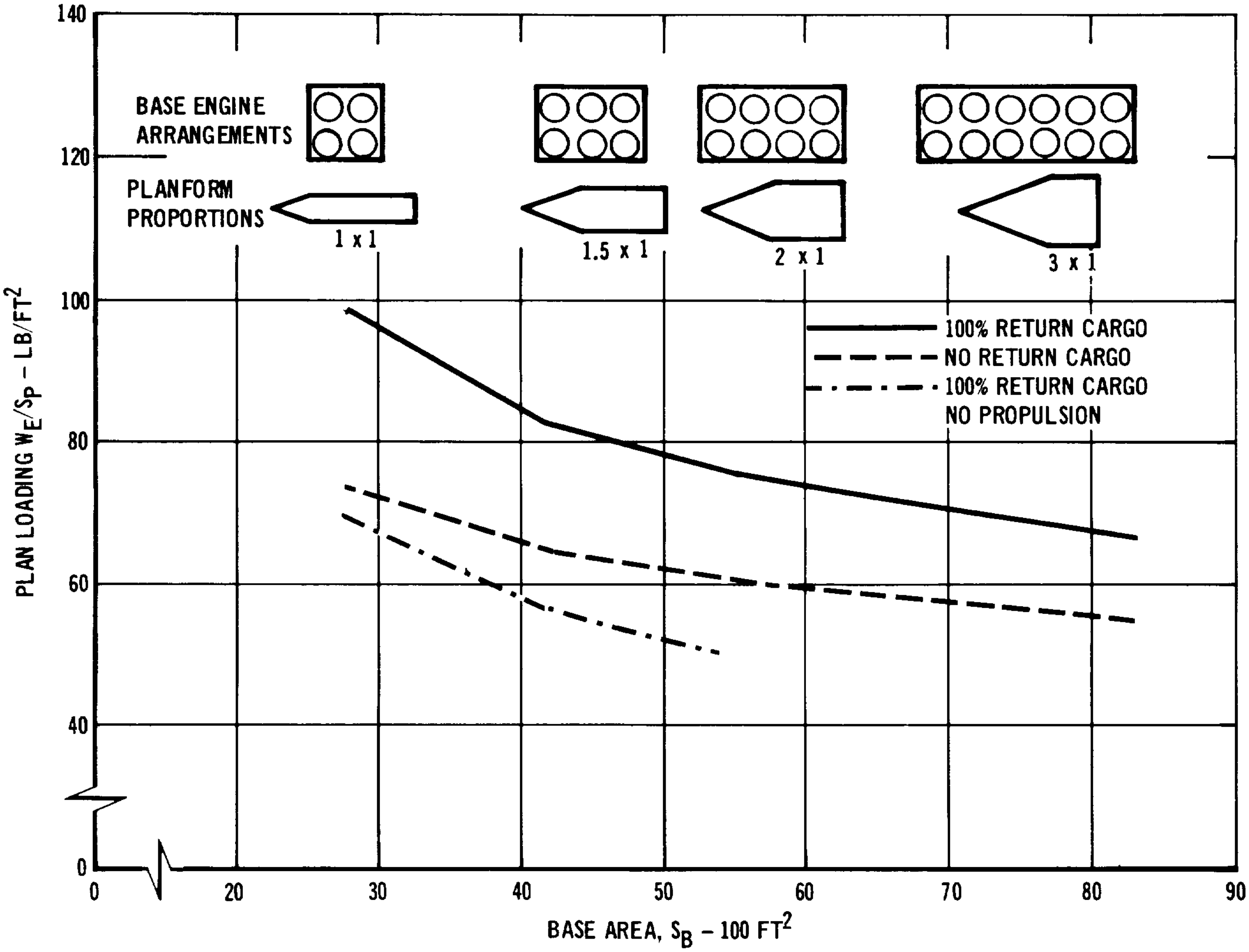


Figure 2-25

plan loading is clearly seen. The reduced slope of the plan loading variation is due to the growth of the propulsion system and to the reduced rate of increase in available internal propellant volume as the span approaches the value for a pure delta. A favorable influence that is accounted for in this data is the higher engine expansion ratio permitted at the higher spans because of the larger base area available for engine installation. This reflects itself in higher vacuum specific impulse values and correspondingly lower total propellant weights (than if the expansion ratio was held constant). As would be expected the rate of decrease in plan loading diminishes with increasing base area.

The effect of scaling on plan loading is shown in Figure 2-26. If the expansion ratio is held constant, the plan loading actually increases, reaches a maximum and then decreases. This is true for both the 100% and the zero return payload cases. If the expansion ratio is allowed to be set at the maximum value for each vehicle size the plan loading does decrease but at a relatively slow rate. This corresponds to about a 10 psf decrease for a 32% increase in vehicle length (100% return cargo). For the 0 return cargo, the plan loading is essentially unaffected by scaling the vehicle. The plan loading for the 2 x 1 vehicle is diminished by scaling up in size but at a lower rate than the 1 x 1 vehicle (100% return cargo). The zero return payload case is similar to the 1 x 1 vehicle. If the propulsion system weight is removed, the plan loadings for both the 1 x 1 and the 2 x 1 vehicle decrease almost linearly with increasing vehicle length.

A point design comparison was made employing the three vehicle configurations shown in Figure 2-27. The vehicle of 130 ft. in length provides for no internal propellant volume and is analyzed in a four-tank configuration similar to that shown in Figures 2-16 and 2-20. This vehicle was scaled to 92 ft. in length and then stretched at constant body cross-section to 130 ft. It represents a minimum envelope for a 15 x 60 cargo bay. The intermediate vehicle of Figure 2-27 was scaled to 160 ft. At this size the body depth was significantly larger than required for the cargo lateral dimentsions. Rather than carry propellant in this upper volume (which did not contribute to plan area) this propellant was assigned to the external tanks and the upper body was sliced off. Propellant volume was used, however, in the body to either side of the cargo bay.

The third vehicle shown in Figure 2-27 was scaled up to 200 ft. It was then sliced in a plane parallel to the bottom similar to the previous vehicle and also cut-off to 160 ft. in length. These vehicles are compared in Tables 2-2 and 2-3,

EFFECT OF SCALING

$\Lambda = 75^0$

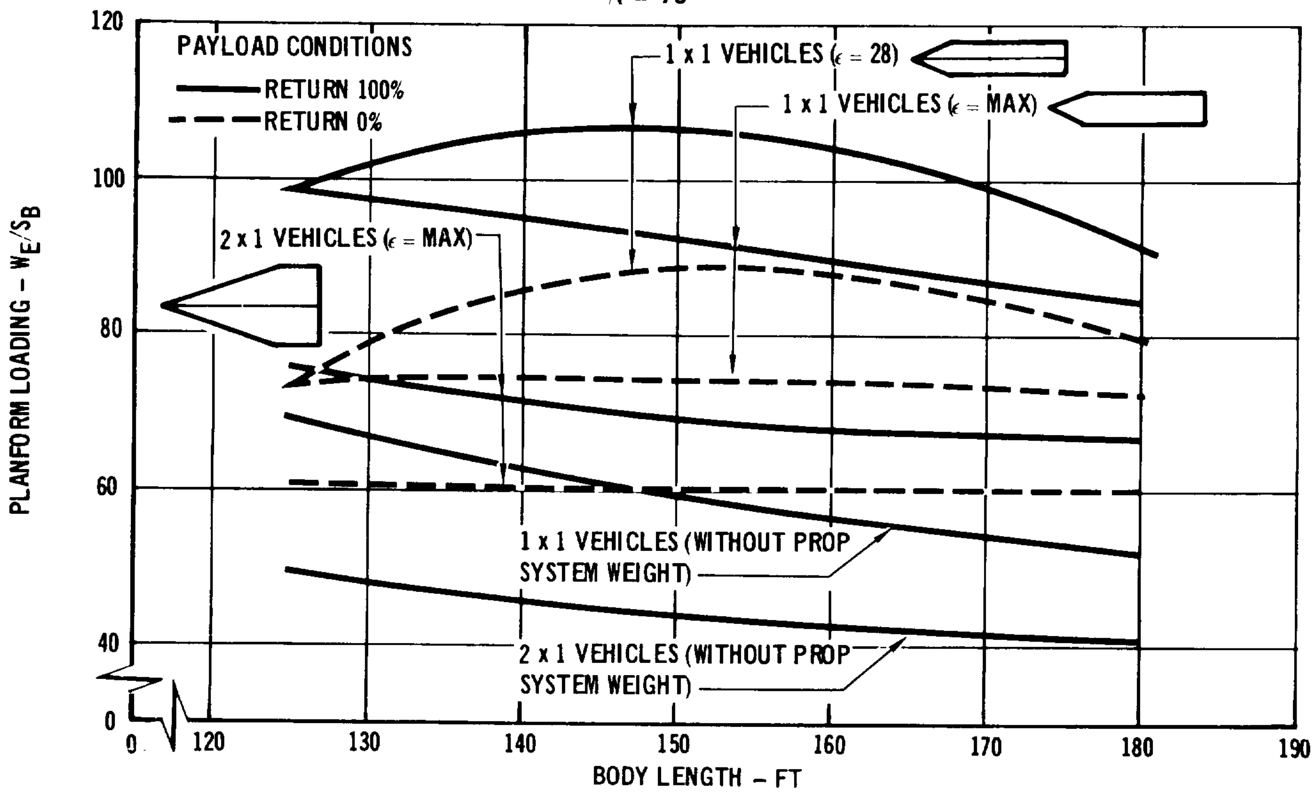


Figure 2-26

CORE VEHICLE SCALING COMPARISON

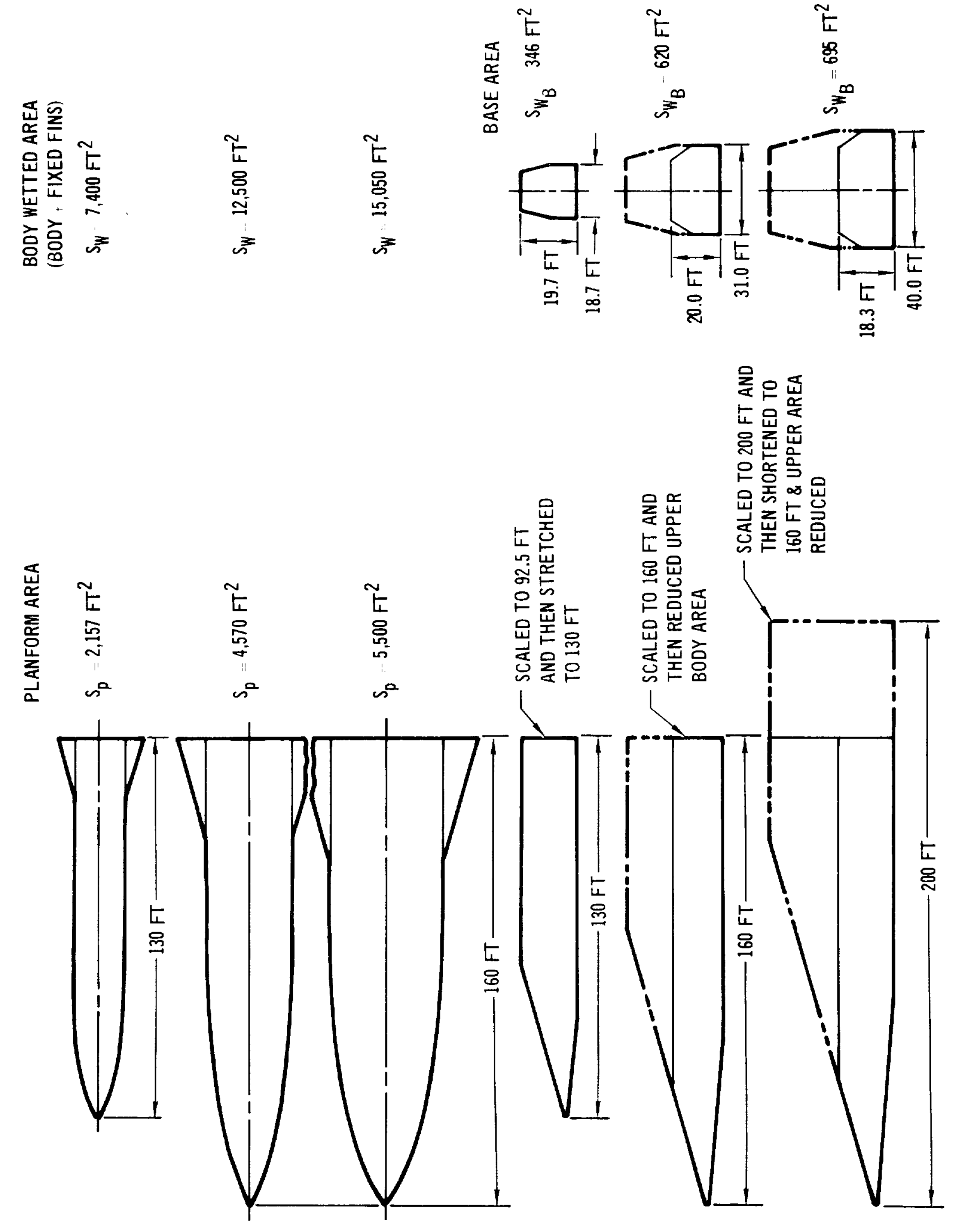


Figure 2–27

Table 2-2

ILRV GENERAL DESCRIPTION
(2 Tip-Tank Configuration Comparison)

CORE VEHICLE LENGTH (FT)	130	160	200/160
CONFIGURATION	–	176M	176M
VARIABLE GEOMETRY WING LOCATION	–	HIGH	HIGH
CREW SIZE	–	3	3
IMPULSIVE VELOCITY CAPABILITY	–	7600	9150
PROPELLANT TYPE	–	O_2H_2	O_2H_2
NUMBER OF ENGINES	–	5	8
CHAMBER PRESSURE (PSI)	–		
SPECIFIC IMPULSE (ISP)	–	422	427
EXPANSION RATIO ϵ	–	20	23.5
TIP TANKS	–	–	–
NUMBER	–	2	2
IMPULSIVE VELOCITY CAPABILITY	–	24,000	22,565
SIDE PAIR	–	24,000	22,565
TOP/BOTTOM PAIR	–	–	–
ILRV DIMENSIONAL DATA			
CORE VEHICLE LENGTH (FT)	130	160	200/160
WETTED AREA (FT^2)	–	–	–
BASIC BODY	–	12,500	15,050
BASE	–	620	695
UPPER TAIL	–	1710	2060
ELEVONS	–	484	580
FLAP	–	540	650
TOTAL	–	15,854	19,035
PLANFORM AREA (BASIC BODY) (FT^2)	–	4570	5500
PLANFORM AREA (WING TO BODY C_L) (FT^2)	–	–	–
LOWER FORWARD RAMP ANGLE (DEGREES)	–	5	5
TIP TANKS	–	–	–
DIAMETER (FT)	–	33	33
LENGTH (FT)	–	204	211

Table 2-3

ILRV GENERAL DESCRIPTION
(4 Tip-Tank Configuration Comparison)

LENGTH CORE VEHICLE (FT)	130	160	200/160
CONFIGURATION	176 M	176 M	176 M
VARIABLE GEOMETRY WING LOCATION	LOW	HIGH	HIGH
CREW SIZE	3	3	3
IMPULSIVE VELOCITY CAPABILITY	0	7960	9720
PROPELLANT TYPE	O_2H_2	O_2H_2	O_2H_2
NUMBER OF ENGINES	4	5	8
CHAMBER PRESSURE (PSI)			
SPECIFIC IMPULSE (ISP)	428	422	433.5
EXPANSION RATIO ϵ	24	20	30.5
TIP TANKS	–	–	–
NUMBER	4	4	4
IMPULSIVE VELOCITY CAPABILITY	31,700	23,760	22,000
SIDE PAIR	14,600	11,880	11,000
TOP/BOTTOM PAIR	17,120	11,880	11,000

ILRV DIMENSIONAL DATA

CORE VEHICLE LENGTH (FT)	130	160	200/160
WETTED AREA (FT^2)	–	–	–
BASIC BODY	7400	12,500	15,050
BASE	346	620	695
UPPER TAIL	805	1710	2060
ELEVONS	228	484	580
FLAP	259	540	650
TOTAL	9038	15,854	19,035
PLANFORM AREA (BASIC BODY) (FT^2)	2157	4570	5500
PLANFORM AREA (WING TO BODY C_L) (FT^2)	–	–	–
LOWER FORWARD RAMP ANGLE (DEGREES)	5	5	5
TIP TANKS	–	–	–
DIAMETER (FT)	21.6	25	25
LENGTH (FT)	153 1ST STAGE 68 2ND STAGE	172 1 ST STAGE 72 2 ND STAGE	181 1 ST STAGE 78 2ND STAGE

in terms of their general and dimensional characteristics and in Table 2-4 in terms of their mass properties. It will be noted that the gross lift off weights increase as do the entry weights of the core vehicle. In going from the 130 ft. vehicle to the 160 ft. scaled vehicle, a decrease in plan loading from 122 to 96 psf is realized. However, an additional increase in plan area in the third vehicle resulted in an increase in entry loading.

The data of Tables 2-5 and 2-6 were analytically derived and describe the characteristics of the 130 ft. and 160 ft. vehicles. The 130 ft. vehicle is a minimum size core vehicle and with its relatively high plan loading of 111 psf employs an ablative lower surface heat shield. The 160 ft. vehicle contains volume for boost-phase propellant and has an entry plan loading of 75.2 psf.

The data of Figure 2-28 summarizes the effect of vehicle size on entry plan loading as a function of engine exapnsion ratio (ϵ) and the lower fixed fin area (S_{LFF}). These data include a point design of a 144 ft. vehicle, a summary of which is not presented, as well as the data for the 130 ft. and 160 ft. vehicles in order to establish the trend shown in Figure 2-28. Employing a lower fin area of 25 percent of the basic body plan area will require a vehicle length of about 165 ft. to reduce the plan loading to approximately 65 psf.

Accordingly, a vehicle was sized at 164 ft, and is illustrated in Figure 2-29 for a four tank arrangement. The gross lift-off weight is 5,713,900 lbs, only slightly different than for the 160 ft. vehicle. The internal arrangement of the 164 ft. vehicle is shown in Figure 2-30. The crew cabin volume for 3 personnel may be easily enlarged to a 12 man capability if required. Boost propellant tanks are located to either side of the cargo bay as well as fore and aft of the bay. Five high chamber pressure engines are employed to provide a total of 7,150,000 lb. of sea level thrust.

The characteristics of the variable geometry wing are illustrated in Figure 2-31. The leading edge sweep angle for this arrangement is 30 degrees, with an exposed area of 684 sq. ft. and an exposed aspect ratio of 6.0. This sketch also illustrates the siamese side propellant tanks.

The geometric characteristics for the 164 ft. vehicle are summarized in Table 2-7. It should be emphasized that the reference plan area used for defining the entry plan loadings does not include the elevon, hypersonic flap or upper tail plan areas. The lower fin characteristics shown in these data are for a fin with 25 percent of the basic body plan area.

Table 2-4

ILRV WEIGHT DATA

(2 Tip Tanks Configuration Comparison)

LENGTH CORE VEHICLE (FT)	130	160	200/160
GROSS WEIGHT AT LIFTOFF	–	7,279,000	8,306,000
CARGO (UP)	–	50,000	50,000
CARGO (DOWN)	–	50,000	50,000
PROPELLANT	–	375,000	581,300
GROSS WEIGHT AT ORBIT INJECTION	–	505,000	619,700
GROSS WEIGHT AT REENTRY	–	474,300	580,780
GROSS WEIGHT AT LANDING	–	451,500	552,580
TIP TANKS	–	–	–
GROSS WEIGHT 1ST PAIR (SIDE)	–	6,404,000	7,105,000
STRUCTURE	–	302,000	335,100
PROPELLANT	–	6,042,000	6,703,000
GROSS WEIGHT 2ND PAIR (TOP/BOTTOM)	–	–	–
STRUCTURE	–	–	–
PROPELLANT	–	–	–

(4 Tip Tanks Configuration Comparison)

LENGTH CORE VEHICLE (FT)	130	160	200/160
GROSS WEIGHT AT LIFTOFF	3,683,000	5,756,000	6,499,000
CARGO (UP)	50,000	50,000	50,000
CARGO (DOWN)	50,000	50,000	50,000
PROPELLANT	–	375,000	581,300
GROSS WEIGHT AT ORBIT INJECTION	280,000	466,000	576,200
GROSS WEIGHT AT REENTRY	262,000	437,000	540,000
GROSS WEIGHT AT LANDING	250,000	416,650	513,800
TIP TANKS	–	–	–
GROSS WEIGHT 1 ST PAIR (SIDE)	2,534,000	3,550,000	3,733,000
STRUCTURE	120,000	167,500	176,100
PROPELLANT	2,400,000	3,350,000	3,522,000
GROSS WEIGHT 2ND PAIR (TOP/BOTTOM)	869,000	1,360,000	1,579,000
STRUCTURE	42,000	64,150	74,480
PROPELLANT	820,000	1,283,000	1,485,000

Table 2-5

ILRV NOMINAL VEHICLE CONFIGURATION CHARACTERISTICS SUMMARY SHEET

CONFIGURATION	50K (15' x 60') (.25 FIXED FIN)	50K (15' x 60') (.25 FIXED FIN)
SPACECRAFT LENGTH (FT) L	130	160
PLANFORM AREA (FT^2) S_p (W/FIXED FINS)	2,713	5,375
BASE AREA (FT^2) S_B	346	620
BODY WETTED AREA (FT^2) S_{WB} (W/FIXED FINS: NO BASE)	8,569	15,374
ELEVON WETTED AREA (FT^2) S_{WE}	165	494
VERTICAL TAIL (BOTH (FT^2) S_{WV}	584	1,746
HYPERSONIC FLAP (FT^2) S_{WF}	230	688
TOTAL WETTED AREA (FT^2) S_{WT}	9,894	18,922
MOLDLINE VOLUME (FT^3) V_m	30,800	57,344
USEABLE VOLUME (FT^3) V_u	24,000	44,654
S/C PROPELLANT VOLUME (FT^3)	0	18,000
S/C ORBIT MANEUVER PROP. VOLUME (FT^3)	2,060	2,760
CARGO (LB):		
DELIVERED	50,000	50,000
RETURNED	50,000	50,000
S/C LIFT-OFF WEIGHT (LB)	356,071	881,091
CORE VEHICLE	356,071	474,291
PROPELLANT	0	406,800
TIP-TANK LAUNCH WEIGHT (LB)	4,708,642	4,829,619
FIRST STAGE STRUCTURE/TANK (LB)	102,397	95,382
PROPELLANT/TANK (LB)	1,759,400	1,638,870
DIAMETER x LENGTH (FT)	25 x 168	23 x 194
SECOND STAGE STRUCTURE/TANK (LB)	54,176	37,430
PROPELLANT/TANK (LB)	465,436	648,125
DIAMETER x LENGTH (FT)	25 x 65	23 x 87
GROSS LAUNCH WEIGHT (LB)	5,064,713	5,710,710
REENTRY WEIGHT (LB)	303,445	406,325
LANDING WEIGHT (LB)	289,745	388,000
PROPULSION SYSTEM:		
PROPELLANT TYPE	O_2/H_2	O_2/H_2
NO. OF ENGINES	4	5
SPECIFIC IMPULSE I_{AP} (SEC)	428	436.5
THRUST (SEA LEVEL) – F_{SL} (LB)	6,328,720	7,135,890

Table 2-6

ILRV NOMINAL VEHICLE CONFIGURATION CHARACTERISTICS – WEIGHT SUMMARY SHEET
(Weight in Pounds)

SPACECRAFT LENGTH (FT)		130*		160*
SCALED LENGTH (FT)		92.3		160
NO. OF MEN		3		3
AERO SURFACES		18,400		57,800
V.G. WING	10,600		30,350	
STRUCTURE		43,500		47,100
THERMAL PROTECTION		31,000		35,300
LANDING + RECOVERY		7,700		13,000
PRIME POWER		3,900		4,600
POWER CONVERSION		1,800		2,100
GUIDANCE + NAVIGATION		420		420
INSTRUMENTATION		200		200
COMMUNICATIONS		220		220
PERSONNEL + PROVISIONS		1,150		1,150
ENVIRONMENTAL CONTROL		1,020		1,020
ORBIT MANEUVER CONTROL SYSTEM		52,084		69,428
INERT	5,208		6,928	
PROP	46,876		62,500	
BOOST ENGINES		87,630		101,673
BOOST FEED SYSTEM		30,333		34,483
GO-AROUND		21,259		28,338
INERT	7,559		10,038	
PROP	13,700		18,300	
CREW STATION CONTROL		260		260
PAYLOAD		50,000		50,000
ORDNANCE + SEPARATION		200		200
ATTITUDE CONTROL SYSTEM		4,995		6,659
CORE VEHICLE PROPELLANT INERTS		0		20,340
CORE VEHICLE PROPELLANT		0		406,800
LAUNCH WEIGHT		356,071		881,091

* CONFIGURATION HAS A .25 Sp FIXED FIN

EFFECT OF VEHICLE SIZE ON ENTRY PLAN LOADING

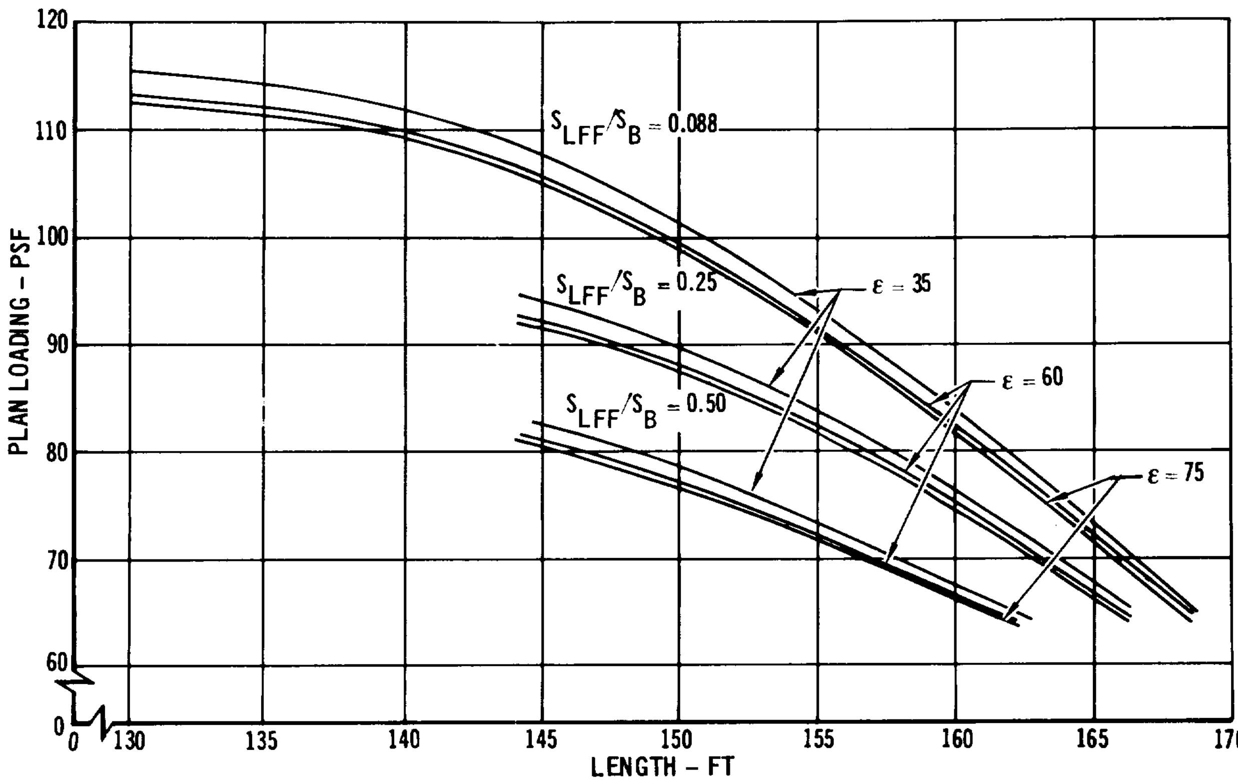
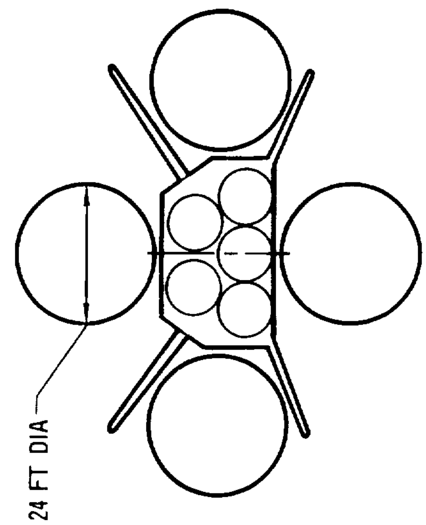


Figure 2-28

LAUNCH CONFIGURATION – ILRV – 164 FT VEHICLE

WEIGHT SUMMARY – LB

CORE VEHICLE	793,900	
PAYLOAD	50,000	
		843,900
TIP TANKS		
1ST STAGE PROPELLANT	3,330,600	3,520,000
2ND STAGE PROPELLANT	1,277,870	1,350,000
TOTAL LAUNCH WEIGHT		5,713,900



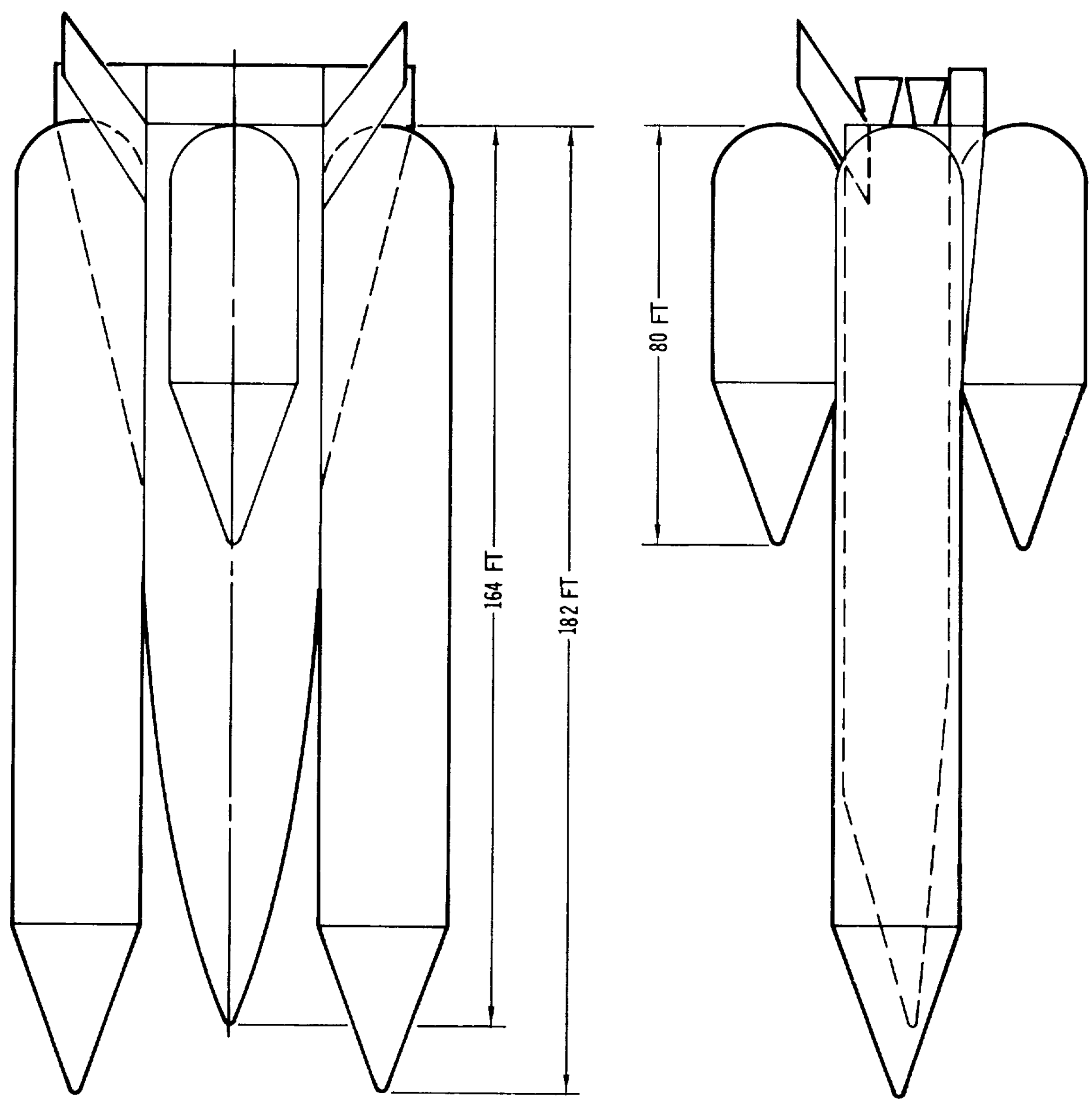


Figure 2–29

GENERAL ARRANGEMENT
(164 Ft. Vehicle: Cargo = 50,000 Lb)

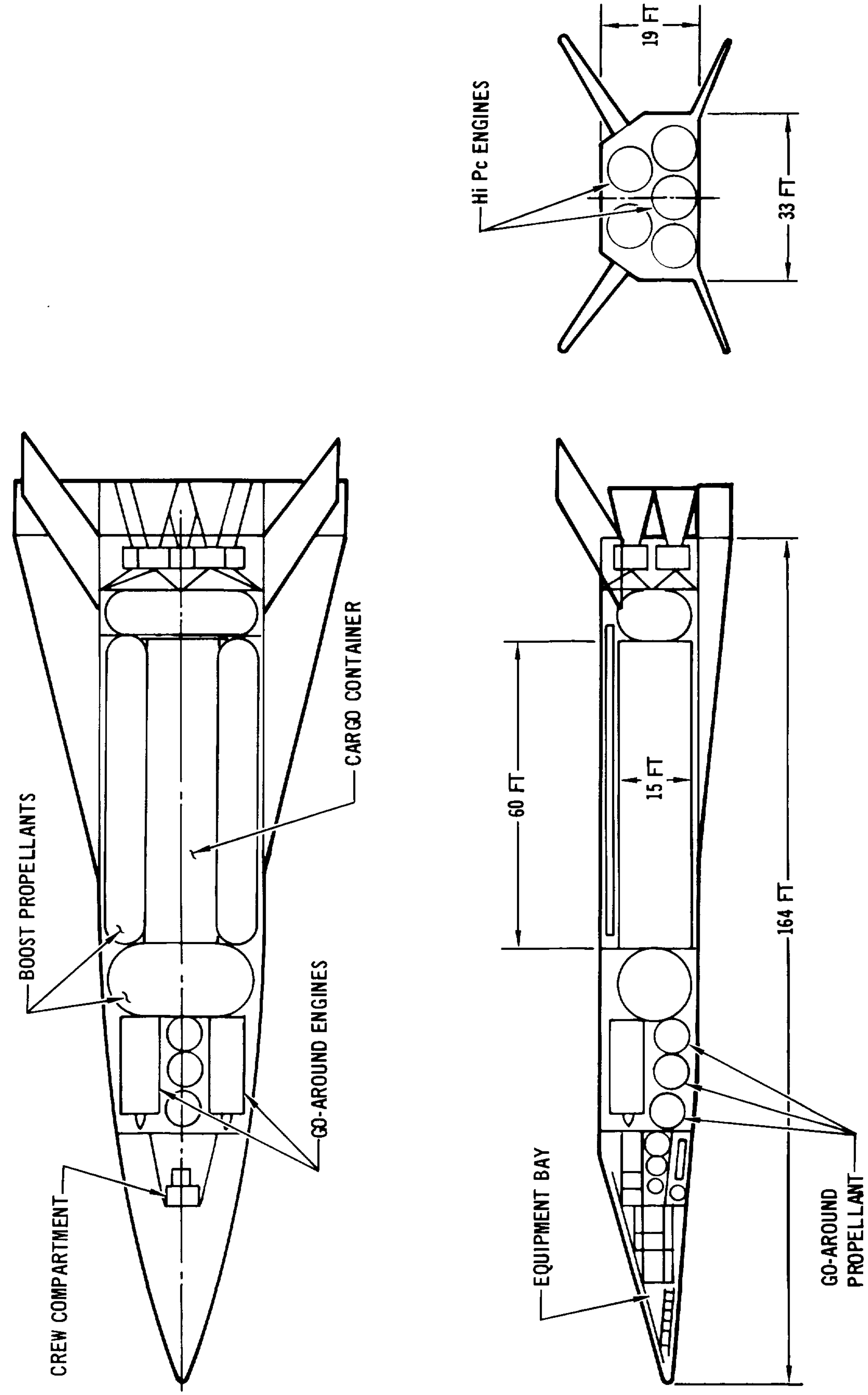


Figure 2–30

WING GEOMETRY
(164 Ft Vehicle)

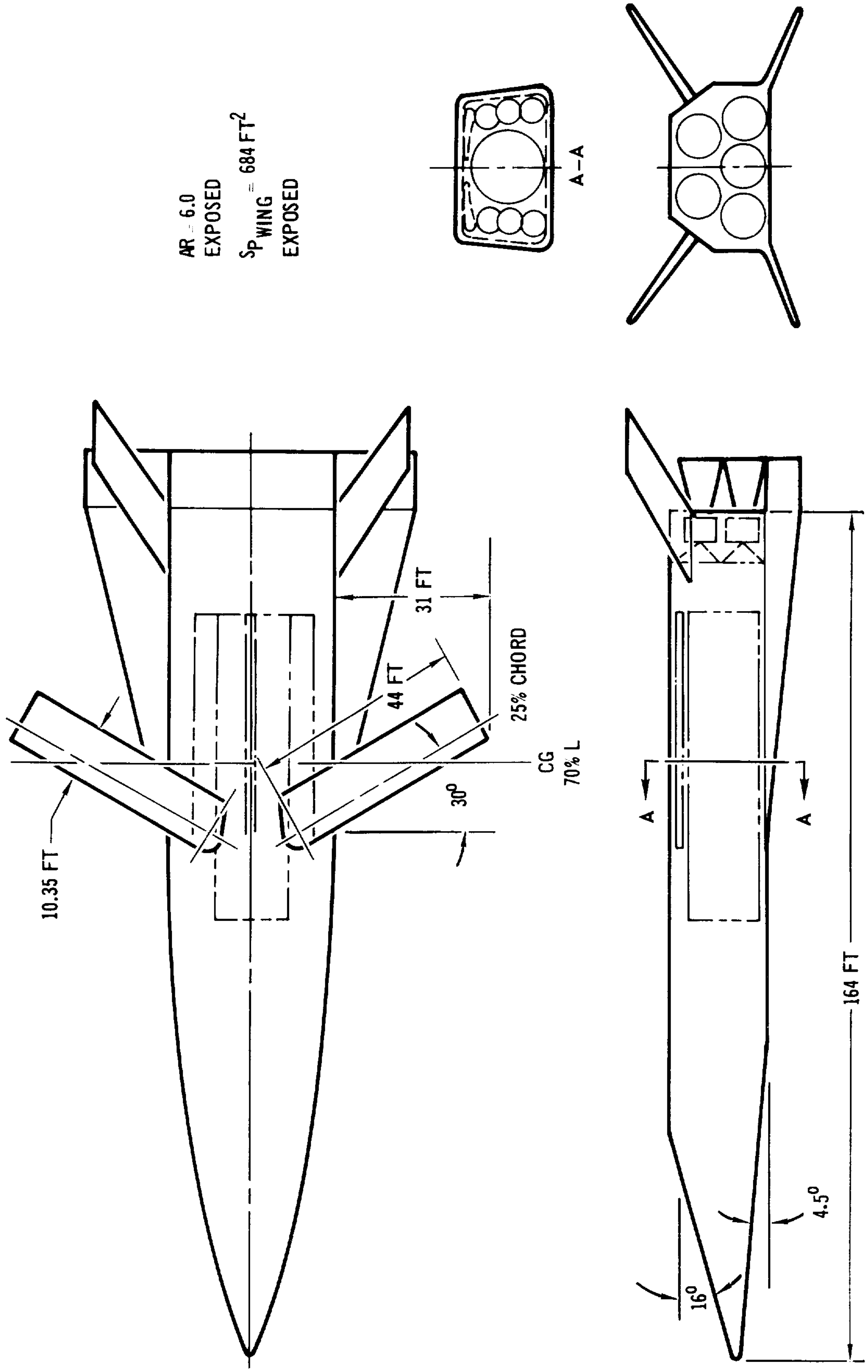


Figure 2–31

Table 2-7

CONFIGURATION CHARACTERISTICS

MODEL		176M
LENGTH, L (FT)		164
CARGO CONTAINER DIMENSIONS, $D_C \times L_C$ (FT)		15 x 60
MAX CARGO WEIGHT, W_C		50,000
DELIVERED TO ORBIT		50,000
RETURNED FROM ORBIT		50,000
REFERENCE PLAN AREA, S_P[1] (FT^2)		5,640
WETTED AREAS		
BASE, S_B (FT^2)		596
BODY, S_{WB} (FT^2)		13,760
LOWER FIXED FIN, S_{WF} (FT^2)		2,260
ELEVON, S_{WE} (FT^2)		790
UPPER TAIL, S_{WT} (FT^2)		1,836
HYPERSONIC FLAP, S_{WH} (FT^2)		720
TOTAL, S_W (FT^2)		19,962
VOLUMES		
BODY MOLDLINE, V_M (FT^3)		74,200
BODY USABLE, V_U (FT^3)		59,500
USABLE PROPELLANT, V_P (FT^3)		19,210
BOOST PHASE, V_{PB} (FT^3)	16,450	
ORBIT PHASE, V_{PO} (FT^3)	2,760	

[1] INCLUDES BASIC BODY (4520 FT^2) PLUS LOWER FIXED FIN (1130 FT^2)

The core vehicle weight distribution is shown in Table 2-8. The thermal protection system is based on the use of re-radiative metallic lower surfaces whose temperatures do not exceed 2500°F. The external tank weight distribution is shown in Table 2-9. The side tanks are used for the initial boost phase and produce about 12,400 fps of theoretical impulsive velocity. The top and bottom pair produce another 12,400 fps of ΔV. The basic usable propellant mass fraction is 0.945 for both pairs of tanks. The propellant distribution between tank pairs can be changed rather easily resulting in changes of the tank lengths. The vehicle weights at various points along the mission profile are summarized in Table 2-10. Also shown are the plan loadings of the initial point of the entry and landing phases.

2.4 Alternate Propellant Distribution - An investigation was conducted based on restricting a given tip tank to a single fluid with a single feed line. This constraint provides a simplified tank and feed system design, as well as minimizing separation complexity.

The alternate distribution for achieving greater simplicity is illustrated in Figure 2-32. The spacecraft general arrangement is quite similar to a conventional propellant distribution previously shown. However, the two side tanks in the alternate arrangement contain hydrogen only. The top centerline tank also contains only hydrogen. The bottom tank contains only liquid oxygen and the core vehicle tankage contains almost all liquid oxygen. The utilization sequence is as follows:

PHASE	PROPELLANT	PROPELLANT SOURCE
1	O_2	Bottom C_L tank
	H_2	Side tanks
2	O_2	Core vehicle
	H_2	Side tanks
3	O_2	Core vehicle
	H_2	Top C_L tank
4*	O_2	Core vehicle
	H_2	Core vehicle

* This phase begins just prior to orbit injection

Table 2-8

CORE VEHICLE – WEIGHT DISTRIBUTION

ITEM	WEIGHT, LB	
AERODYNAMIC SURFACES		45,450
EMPENNAGE	29,700	
VARIABLE GEOMETRY WING	15,750	
STRUCTURE		54,990
THERMAL PROTECTION		26,820
LANDING		10,170
PRIME POWER		4,140
POWER CONVERSION		1,890
GUIDANCE AND NAVIGATION		378
INSTRUMENTATION		180
COMMUNICATION		198
ENVIRONMENTAL CONTROL		918
CREW STATION CONTROL		234
ORDNANCE AND SEPARATION		180
BOOST PROPULSION		511,873
ENGINES AND INSTALLATION	92,218	
FEED SYSTEM	31,079	
TANKAGE	13,371	
RESIDUAL PROPELLANTS	3,751	
USABLE PROPELLANTS	371,454	
ORBIT MANEUVER PROPULSION		68,509
ENGINE AND TANKAGE	5,663	
RESIDUAL PROPELLANTS	622	
USABLE PROPELLANTS	62,224	
ATTITUDE CONTROL SYSTEM		6,524
INERT	964	
RESIDUAL PROPELLANTS	250	
USABLE PROPELLANTS	5,310	
LANDING PROPULSION SYSTEM		27,376
ENGINES AND TANKAGE	7,586	
RESIDUAL PROPELLANT	940	
USABLE PROPELLANT	18,850	
CONTINGENCY ALLOWANCE		32,956
PERSONNEL AND PROVISIONS		1,150
PAYLOAD		50,000
LAUNCH WEIGHT		843,900

Table 2-9

TANK WEIGHT DISTRIBUTION

	(WT., LB.)
FIRST STAGE (SIDE TANK PAIR), EACH TANK	
TANK STRUCTURE AND PRESSURIZATION	77,736
RESIDUAL PROPELLANTS	8,327
USABLE PROPELLANTS	1,665,300
INERT CONTINGENCY	8,637
LAUNCH WEIGHT	1,760,000
SECOND STAGE (TOP/BOTTOM TANK PAIR), EACH TANK	
TANK STRUCTURE AND PRESSURIZATION	29,583
RESIDUAL PROPELLANTS	3,195
USABLE PROPELLANTS	638,935
INERT CONTINGENCY	3,287
LAUNCH WEIGHT	675,000

Table 2-10

MISSION WEIGHT AND AREA LOADING SUMMARY

	(WT.,LB)	(W/S)
LAUNCH	5,713,900	
1ST TIP TANK STAGE BURNOUT	2,383,300	
2ND TIP TANK STAGE INITIAL	2,193,900	
2ND TIP TANK STAGE BURNOUT	916,030	
INITIATION OF FINAL BOOST PHASE	843,900	
ORBIT INJECTION	472,446	
ENTRY	404,900	
PLAN LOADING, PSF		71.5
LANDING INITIATION	386,050	
PLAN LOADING V G WING DEPLOYED, PSF		61.0
PLAN LOADING, V G WING STOWED, PSF		68.3

PROPELLANT DISTRIBUTIONS

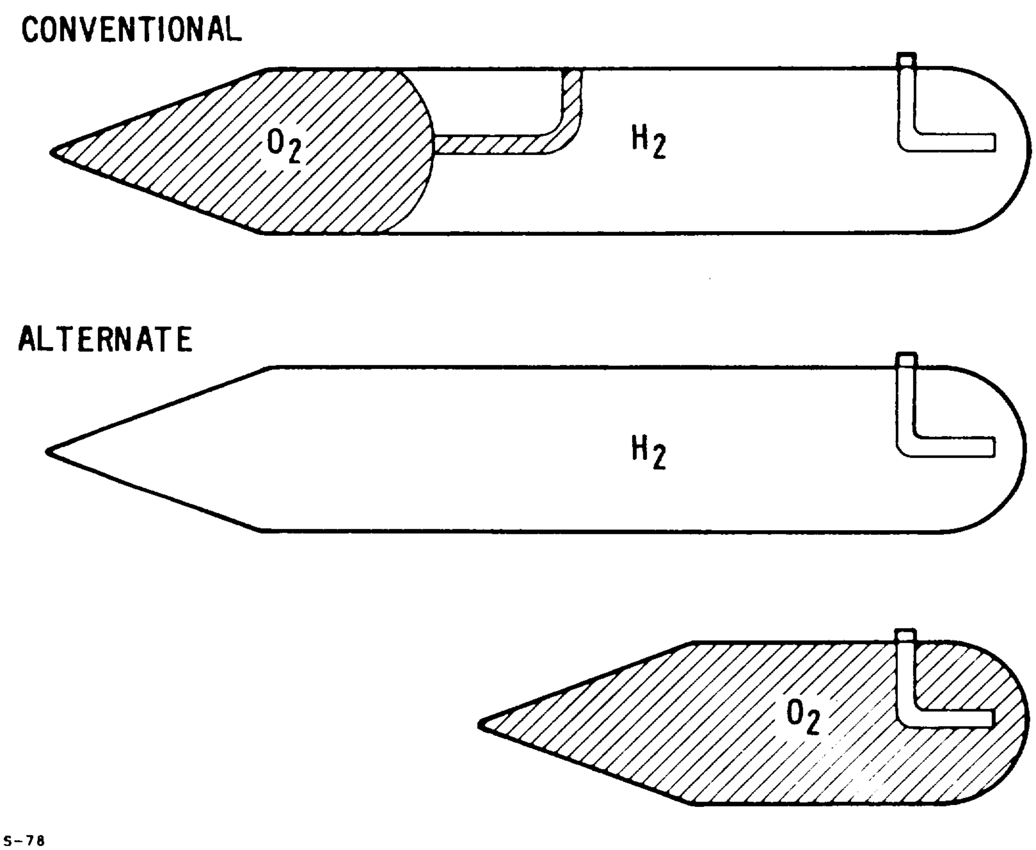


Figure 2-32

Each tip tank now has only one feed line connection to the core vehicle. There are no common bulkheads and the tank inertia loads transmitted to the core vehicle are significantly smaller.

A weight summary for this vehicle is presented in Table 2-11. A weight summary for the 1 1/2 stage concept, with 50,000 lb cargo in a 15 ft diameter 60 ft long container, is shown in Table 2-12.

Table 2-11

ONE AND A HALF STAGE WEIGHT SUMMARY

(Weight in Pounds)

LENGTH – FT	130	130
TANK ARRANGEMENT	ALTERNATE	CONVENTIONAL
AERODYNAMIC SURFACES	35,300	35,300
BODY STRUCTURE	45,700	45,700
THERMAL PROTECTION	38,600	38,600
LANDING SYSTEM	8,000	8,000
MAIN PROPULSION SYSTEM	1,013,100	353,200
SECONDARY PROPULSION SYSTEM	49,300	46,400
ENTRY ATTITUDE CONTROL SYSTEM	4,700	4,600
LANDING PROPULSION SYSTEM	20,100	19,400
SUBSYSTEMS AND CREW	9,540	9,540
CARGO	25,000	25,000
CORE LAUNCH WEIGHT	1,249,340	585,740
TIP TANK SYSTEM	1,822,300	2,442,600
GROSS LAUNCH WEIGHT	3,071,640	3,028,340

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Table 2-12

ONE AND A HALF STAGE WEIGHT SUMMARY

(Weight in Pounds)

LENGTH - FT	160	160
TANK ARRANGEMENT	ALTERNATE	CONVENTIONAL
AERODYNAMIC SURFACES	50,200	50,200
BODY STRUCTURE	61,100	61,100
THERMAL PROTECTION	51,500	51,500
LANDING SYSTEM	11,300	11,300
MAIN PROPULSION SYSTEM	2,316,500	784,600
SECONDARY PROPULSION SYSTEM	74,800	69,000
ENTRY ATTITUDE CONTROL SYSTEM	7,200	6,800
LANDING PROPULSION SYSTEM	30,500	28,900
SUBSYSTEMS AND CREW	10,940	10,940
CARGO	50,000	50,000
CORE LAUNCH WEIGHT	2,664,040	1,124,340
TIP TANK SYSTEM	2,085,500	3,333,200
GROSS LAUNCH SYSTEM	4,749,540	4,457,540

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2.5 Configuration Description (25K - 15 ft x 30 ft Payload) - The external characteristics of a 1 1/2 stage vehicle with a 25,000 lb payload contained in a 15 ft diameter by 30 ft length cylindrical cannister in the launch configuration is illustrated in Figure 2-33. The four drop tanks each contain both liquid oxygen and liquid hydrogen separated by a common bulkhead. Volume distributions within each tank correspond to a mixture ratio of 6:1 plus allowances for ullage. The two side tanks are designed to produce about one-third of the theoretical boost impulsive velocity requirements. The second stage of boost energy is furnished from the top and bottom centerline tanks. This increment in impulsive velocity is equal to that of the first stage. The final stage of boost energy is provided from core vehicle internal tankage. In this verison of the launch configuration, all drop tanks are 21.6 ft in diameter and the tank pairs are equal length. The side tanks are each 115 ft long or 15 ft shorter than the basic body of the core vehicle. The length of the centerline tanks was selected to permit the swinging of a nose section through a 90° displacement, shown in Figure 2-34, during prelaunch preparations. The drop tank nose configuration is a 20-deg. half angle cone with a 1.75 ft radius. The aft end of the tanks is formed by a hemispherical section. Volumetric utilization is almost 100 percent.

Each tank is supported at two fore and aft locations. A yoke structure carries a part of the thrust loads into the tank at the juncture of the hemispherical aft dome and also serves as a pivot point during the tank jettison operation. The forward tank attachemnt reacts tank sheer loads only, while the aft attachment carries shear loads, axial loads and torques. During jettison operation, the forward support point is released as jettison rockets in the tank fire to rotate the tank outward and aft of the core vehicle. Support and jettison of the two centerline tanks is similar to the side tank.

The core vehicle geometry and internal arrangement is shown in Figure 2-35. Since the center of pressure of the baseline planform shape is relatively far forward, more than adequate space is available for storing and deploying a variable geometry wing with a forward pibot and small sweep angle. This characteristic permits consideration of wing planforms with reasonably good efficiency. One of the design features of the core vehicle is the integration of the payload. The dominating features are the payload dimensions: 30 ft long with a diameter of 15 ft, a density of about 5 lb/ft^3 and a requirement for deployment outside of the core vehicle while in orbit. In addition, lacking any specifics on the

GENERAL ARRANGEMENT LAUNCH CONFIGURATION

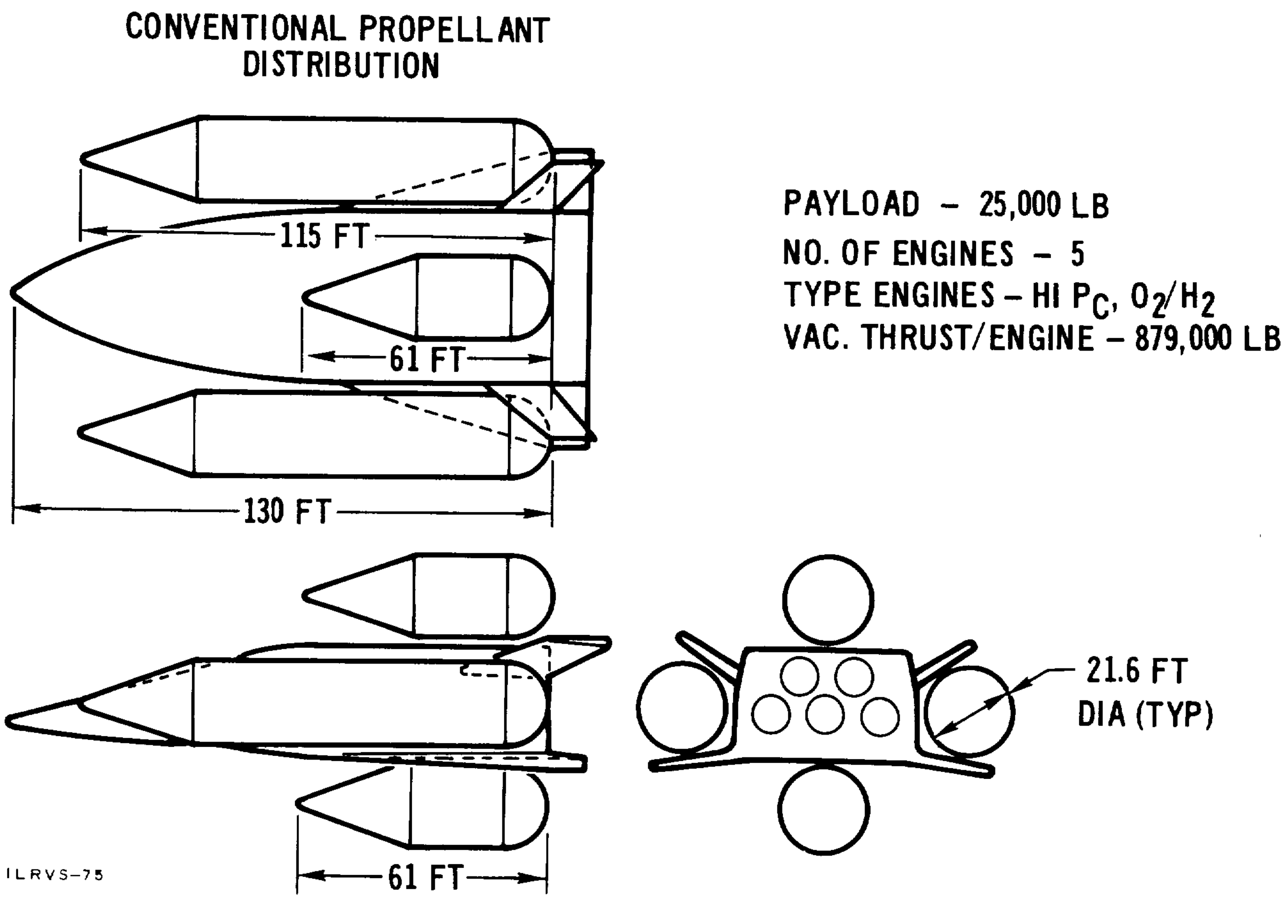
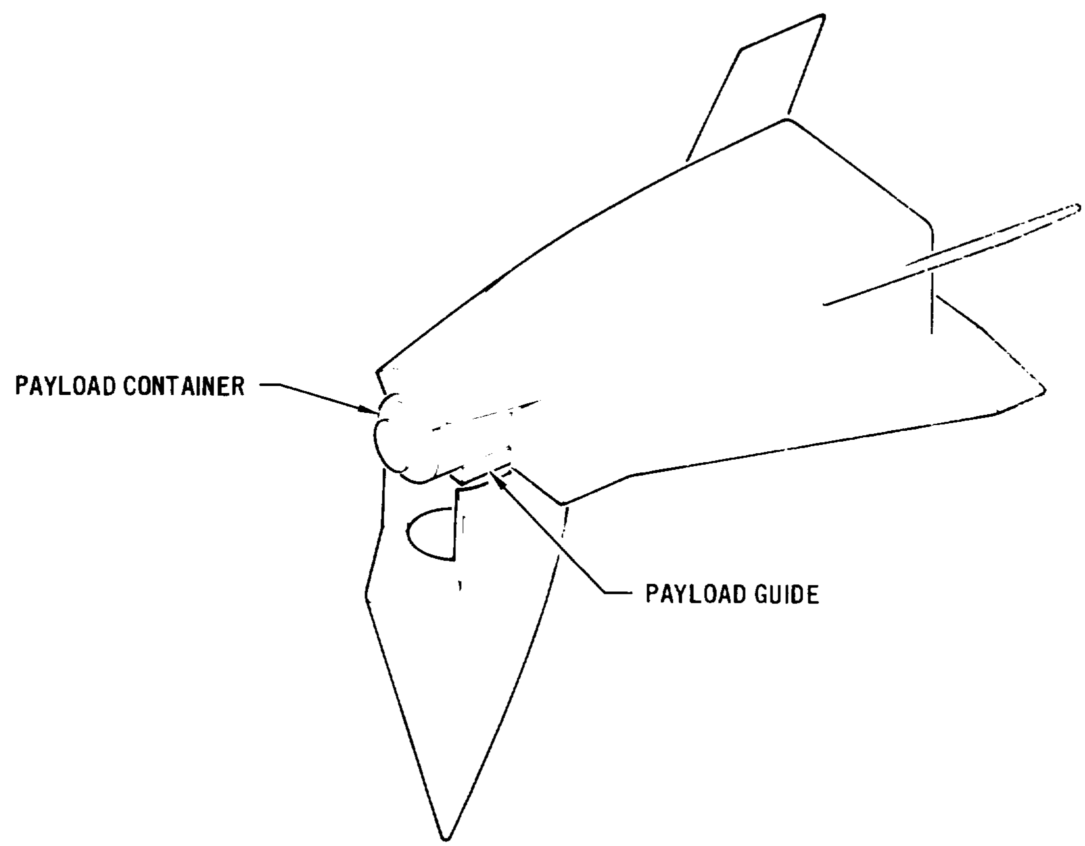


Figure 2-33

center of gravity, the payload mass is assumed to be homogenous, and the requirement for return from orbit with or without the cargo requires an alignment of the payload center of mass with the desired gross vehicle CG.

The payload, in this arrangement, is supported on three longitudinal rails which are joined at the aft end of cargo bay area in a yoke arrangement to provide end restraint for the payload container. Removal of the payload or the translation of the payload for deployment of internal devices is accomplished by rotating the nose section as shown in Figure 2-34 around an articulate hinge located just forward of the cargo bay. In the baseline arrangement, the crew cabin, most of the subsystems, the nose gear, and the landing propulsion system are located in the nose section. The crew cabin is sized for three crewmembers and is located at about the midpoint of the nose. The landing propulsion system is located just aft of the crew cabin and consists of two JP-4 fueled turbojets. The turbojets are mounted in pod-type nacelles and in subsonic flight are rotated outward on the stub wing-type of strut. Propellants are carried in three tank elements. The engine bay begins at the aft end of the tank elements and houses five high pressure bell engines. The variable geometry wing is installed under the cargo bay. This installation is illustrated in Figure 2-35, and deployed positions are shown in Figure 2-36.

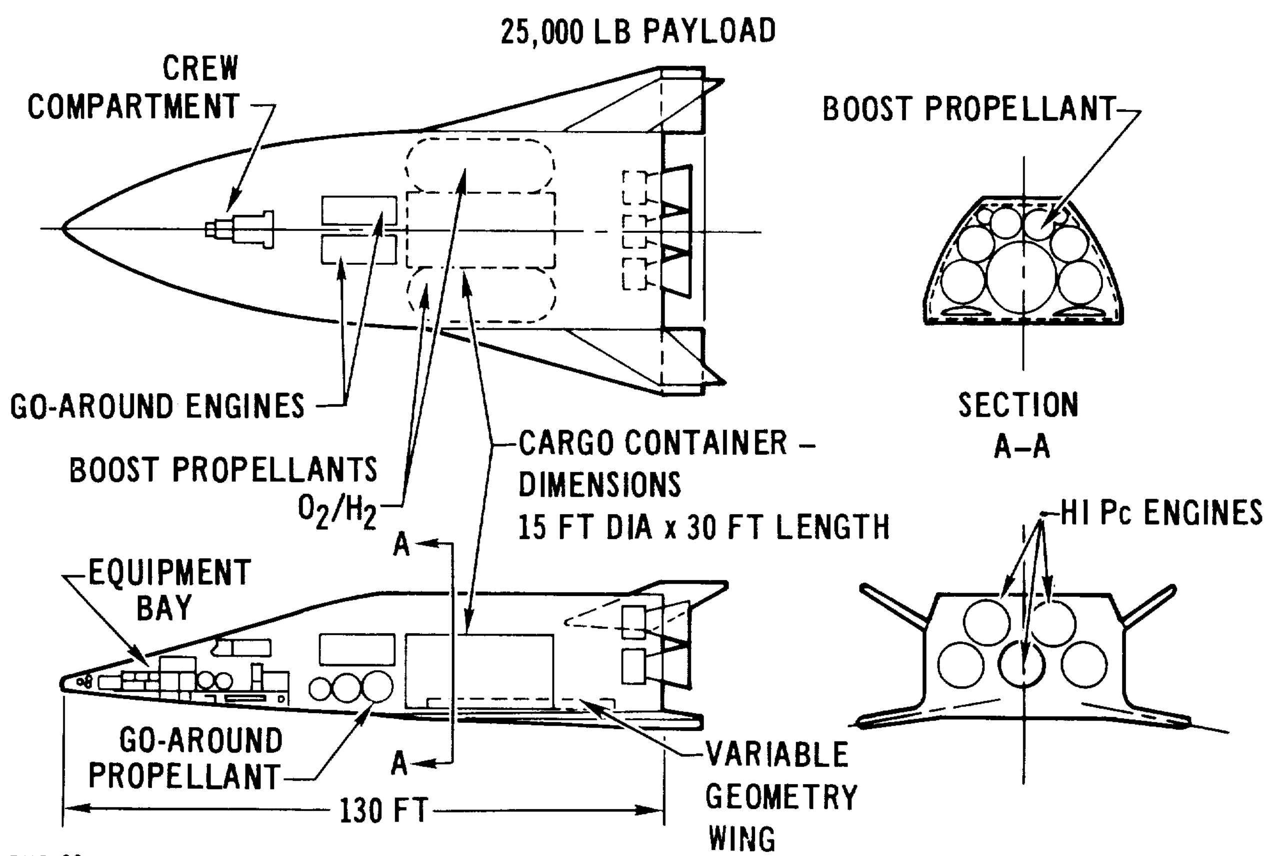
ORBIT CONFIGURATION CARGO DEVELOPMENT MODE
130 Ft Vehicle; Cargo = 50,000 Lb



ILRVS-67

Figure 2-34

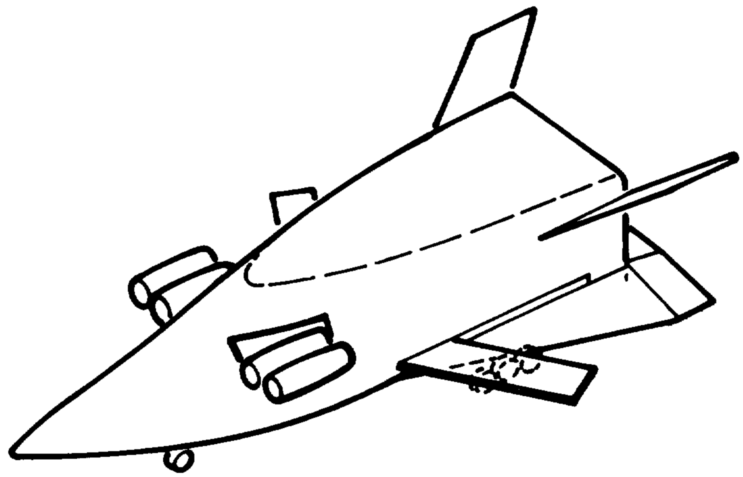
GENERAL ARRANGEMENT – ORBIT AND ENTRY



ILRVS-69

Figure 2–35

WEIGHT SUMMARY – LANDING CONFIGURATION
Conventional Propellant Distribution – 130 Ft Vehicle



LANDING INITIATION
(FULL RETURN CARGO)

V.G. WING DEPLOYED	
WEIGHT (LB)	265,709
PLAN LOADING(PSF)	44
V.G. WING STOWED	
PLAN LOADING(PSF)	50

3.0 PERFORMANCE ANALYSES

These performance analyses include the aerodynamic characteristics for three different length vehicles (67.2, 81.5 and 95.3 feet) with an analytical study of the effects of Mach number on vehicle characteristics for the 81.5 foot vehicle; trajectory analyses for the four tip-tank configuration considering trajectory shaping, payload sensitibity to sizing parameters and inert weight uncertainites, and effects of area loading and loft coefficient and reentry profiles; and the effect of impulsive velocity, orbital inclination and orbit altitude on payload capability.

3.1 Aerodynamic Characteristics - Because of choice of control modes was not made, the comparisons presented in the following figures are made for the non-trimmed condition of zero. Figure 3-1 shows the lift/drag characteristics for three different length vehicles. These are complete vehicles with all surfaces included. Each of these vehicles has the same nose and the same after-body, with a constant cross-section length spliced in the middle. The peak value of lift/drag is presented in Figure 3-2 as a function of vehicle length. As the vehicle length increases the lift/drag value increases over the range of lengths studied. This increase is due to the addition of surfaces contributing largely to normal force, while creating small changes in axial force.

It was necessary to obtain some idea of the changes caused by Mach number variation. Since the final vehicle length had not been defined, the 81.5 foot configuration was selected to be representative for the analytical investigation of Mach number effects in which viscous effects are included. The lift/drag characteristics for three Mach numbers are shown in Figure 3-3. The peaks of these curves, as well as the corresponding angle of attack and lift coefficient are shown in Figure 3-4 as a function of Mach number.

MODEL 176 M – EFFECT OF LENGTH ON L/D
MACH 20 h = 200,000 Ft
Zero Control Deflections

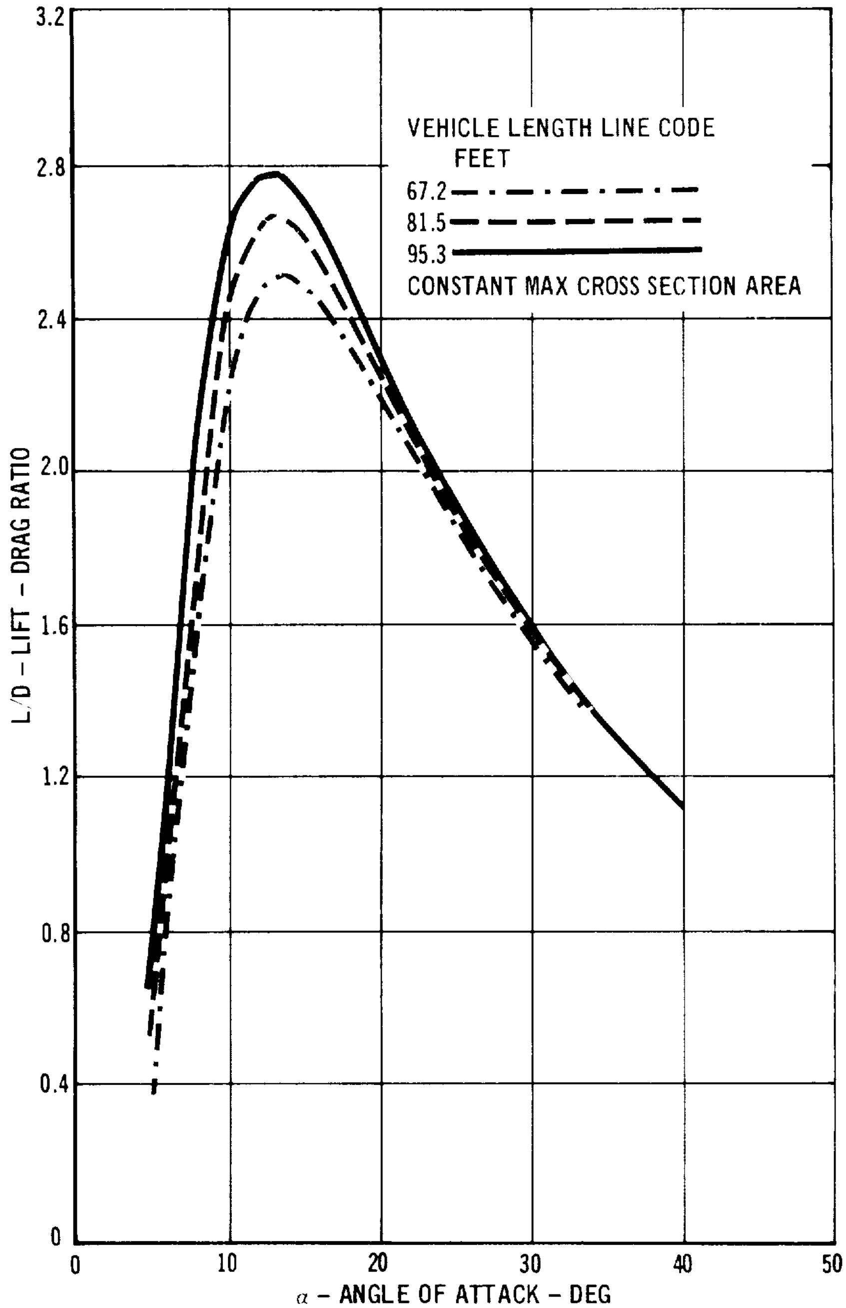


Figure 3-1

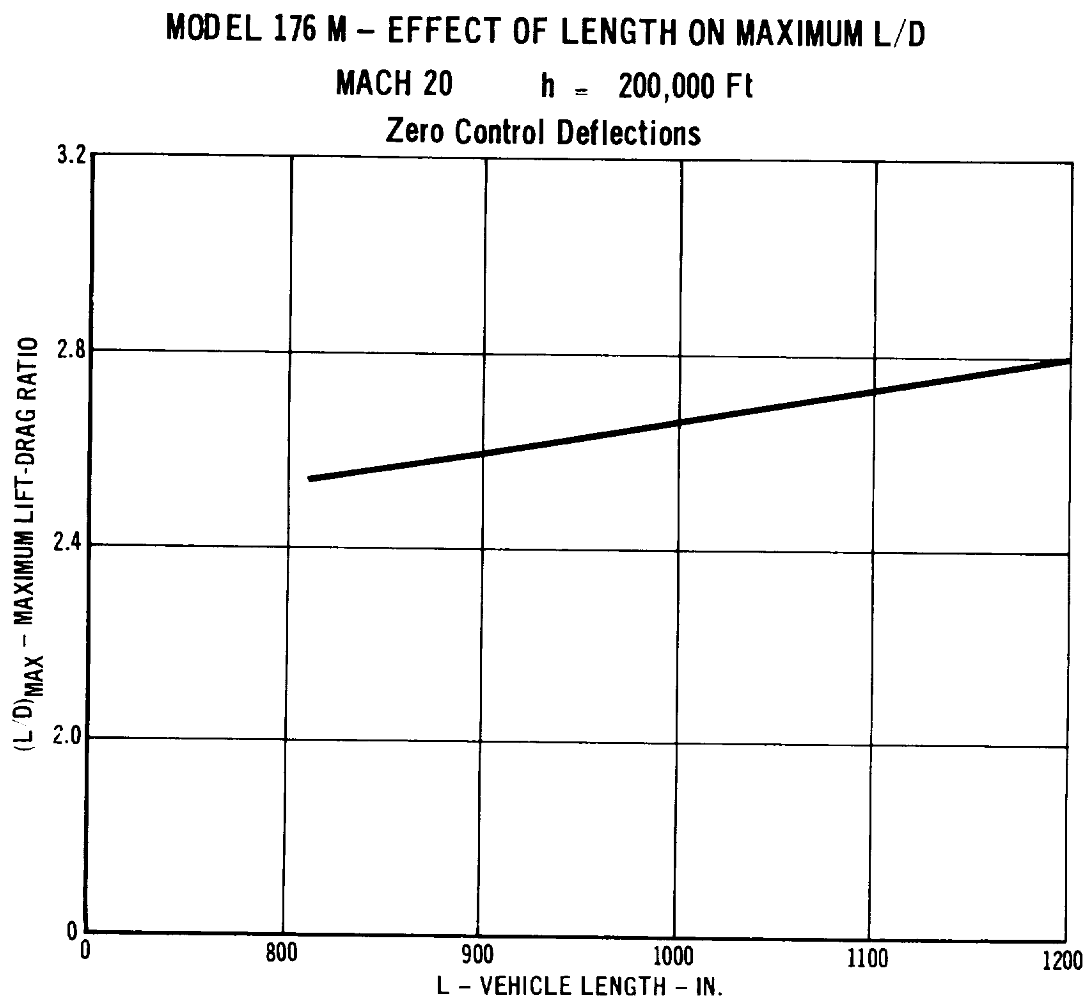

MODEL 176 M - EFFECT OF LENGTH ON MAXIMUM L/D
MACH 20 h = 200,000 Ft
Zero Control Deflections
3.2
2.8
2.4
2.0
0
(L/D)MAX - MAXIMUM LIFT-DRAG RATIO
0
800
900
1000
1100
1200
L - VEHICLE LENGTH - IN.

Figure 3-2

MODEL 176M – MACH EFFECT ON L/D

1000 Inch Vehicle
Zero Control Deflections

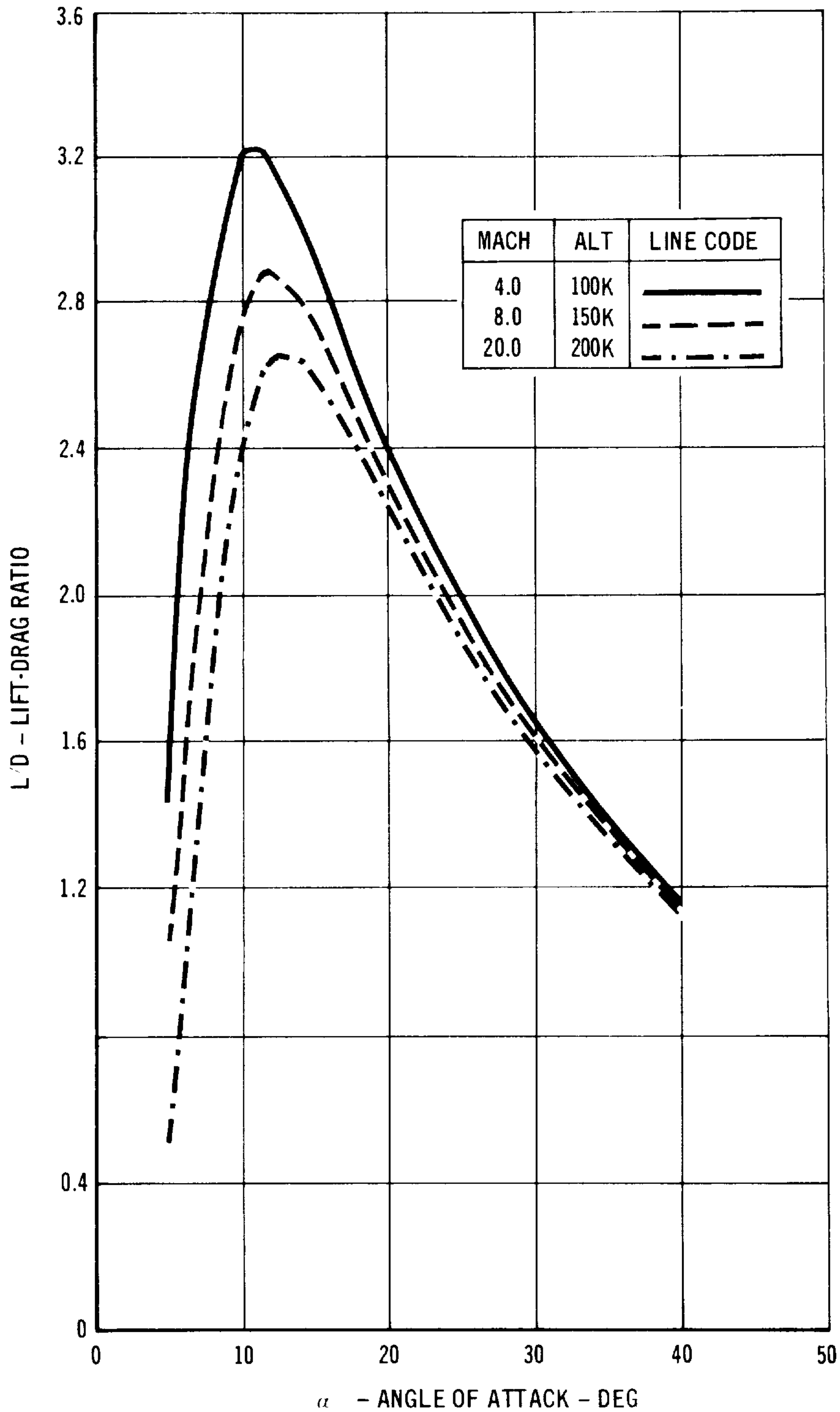


Figure 3–3

MODEL 176M – MACH EFFECT ON OPTIMUM CHARACTERISTICS

1000 Inch Vehicle Zero Control Deflections

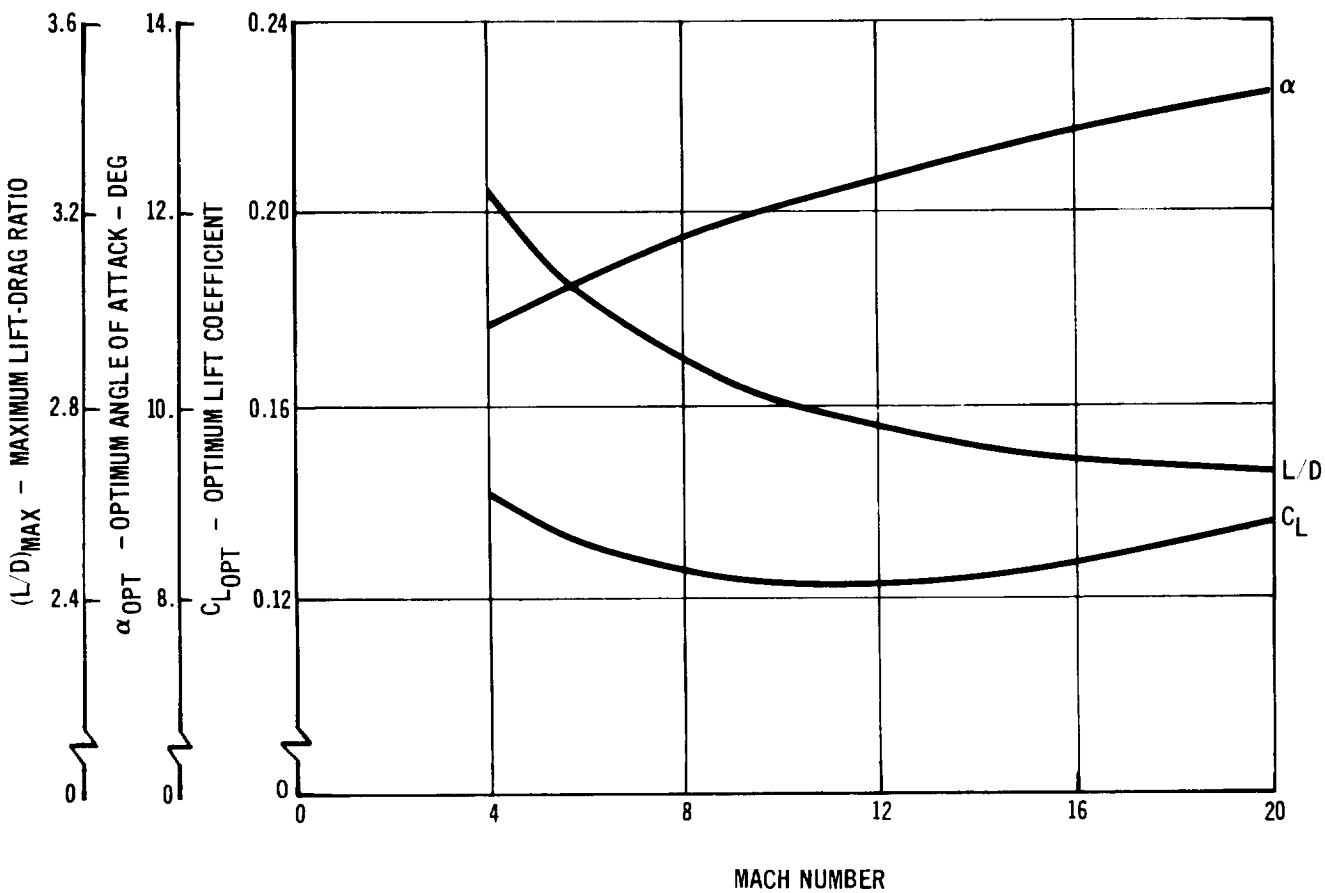


Figure 3–4

3.2 Trajectories and Performance - Detailed trajectory analyses were initiated for the 1 1/2 stage, 4 tip-tank 130 foot long configuration. Powered ascent-phase and unpowered reentry and glide phase simulations were performed. The subjects which were addressed and are reported here were:

a) Ascent-phase trajectory shaping
b) Payload capability of the spacecraft configured for a 15 ft. x 60 ft. 50,000 lb. cargo
c) Effects of limiting axial acceleration to 3 g.
d) Sensitivity of payload capability to principal sizing parameter and subsystem weight uncertainties, and
e) The effects of area loading (W/S) and lift coefficient (C_L) on the reentry and glide trajectory.

3.2.1 Ascent Phase Shaping - The proposed nominal mission trajectory profile consists of injection at perigee of an elliptic orbit with apogee at 100 na mi. Preliminary results indicated that velocity losses due to gravity, and thrust vectoring could be substantially reduced by targeting injection to an altitude of 65.83 na mi compared with direct injection at 100 na mi. The flight plan consists of a vertical rise for approximately 40 seconds followed by a rapid pitch maneuver of 4.56 degrees. After 120 seconds of gravity turn flight, optimization of thrust deflection was initiated to attain desired injection conditions with minimum attendant velocity losses. The thrust level was modulated to maintain a maximum thrust-to-weight ratio of approximately 4 g. At injection there were 32,890 lbs. of fuel remaining. The requirements for circularization at 100 na mi and orbital maneuvers amount to 22,234 lbs. leaving an excess of approximately 10,656 lbs. which could be considered as equivalent excess payload. This performance gain resulted from the trajectory shaping which suppressed and ascent-phase velocity losses to less than that initially budgeted for spacecraft sizing.

Several pertinent trajectory parameters are presented as a function of time from lift-off in Figures 3-5 and 3-6 A breakdown of velocity losses is presented in Figure 3-7.

The subject trajectory was recomputed with a 3 g limit on maximum thrust-to-weight ratio. The result was an additional one percent improvement in payload (i.e., 11,450 lbs.) An instinctive conclusion is that lower g's improve performance and that further reduction in maximum g's would continue to improve

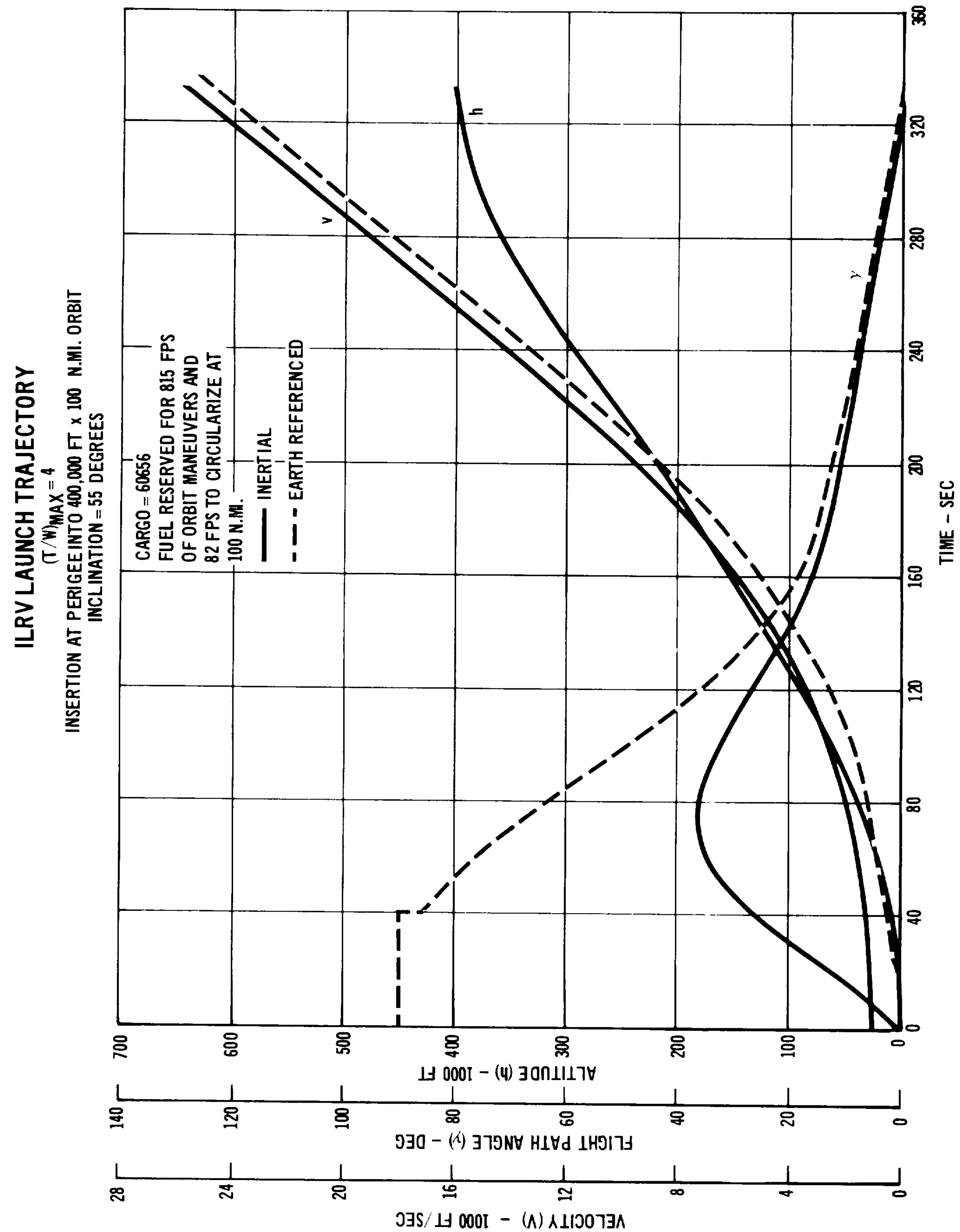

ILRV LAUNCH TRAJECTORY
(T/W)MAX = 4
INSERTION AT PERIGEE INTO 400,000 FT x 100 N.MI. ORBIT
INCLINATION = 55 DEGREES
CARGO = 60656
FUEL RESERVED FOR 815 FPS OF ORBIT MANEUVERS AND 82 FPS TO CIRCULARIZE AT 100 N.MI.
INERTIAL
EARTH REFERENCED
h
V
γ
TIME - SEC
0
40
80
120
160
200
240
280
320
360
ALTITUDE (h) - 1000 FT
0
100
200
300
400
500
600
700
FLIGHT PATH ANGLE (γ) - DEG
0
20
40
60
80
100
120
140
VELOCITY (V) - 1000 FT/SEC
0
4
8
12
16
20
24
28

Figure 3-5

ILRV LAUNCH TRAJECTORY

$(T/W)_{MAX} = 4$

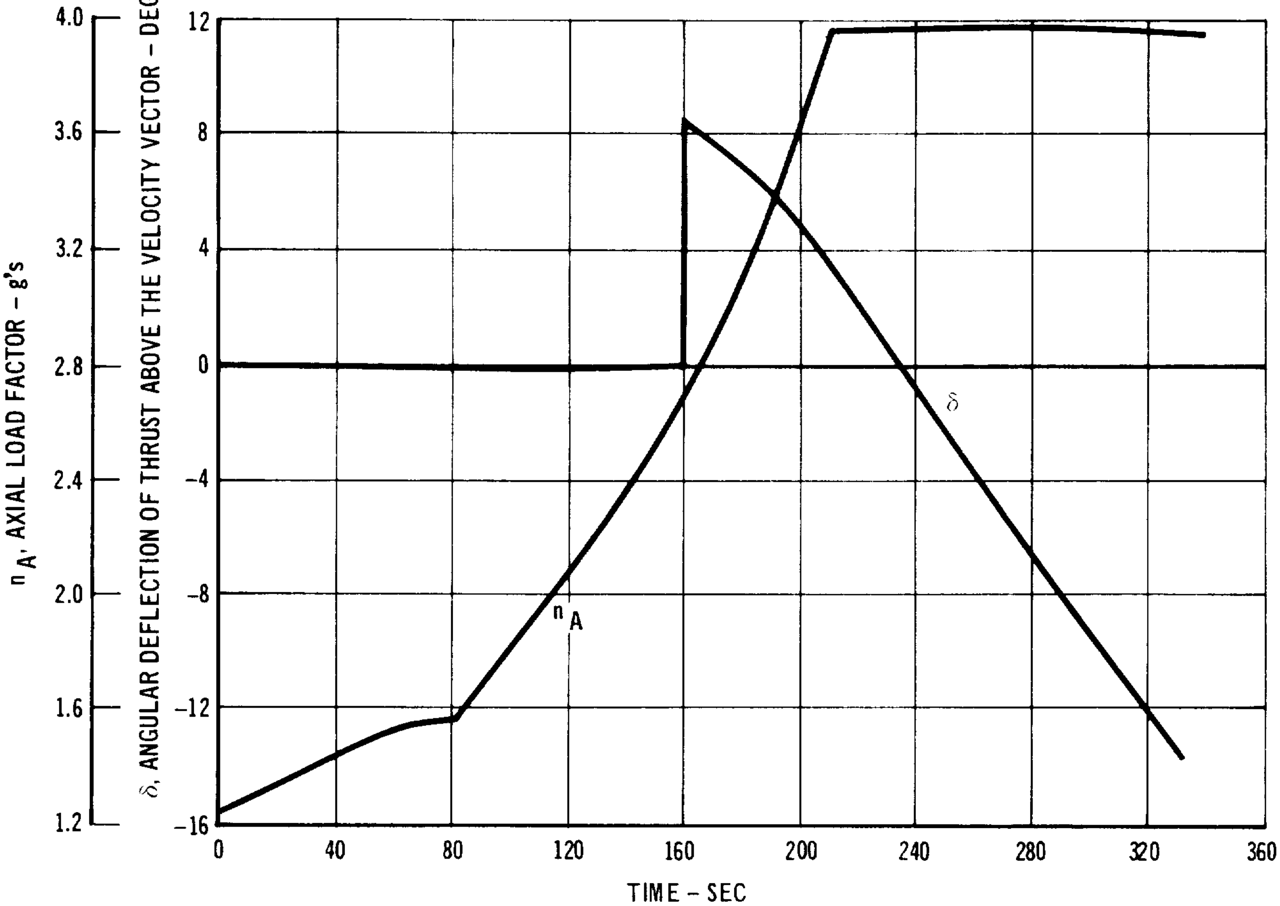


Figure 3-6

VELOCITY LOSSES FOR ILRV LAUNCH TRAJECTORY

$(T/W)_{MAX} = 3$

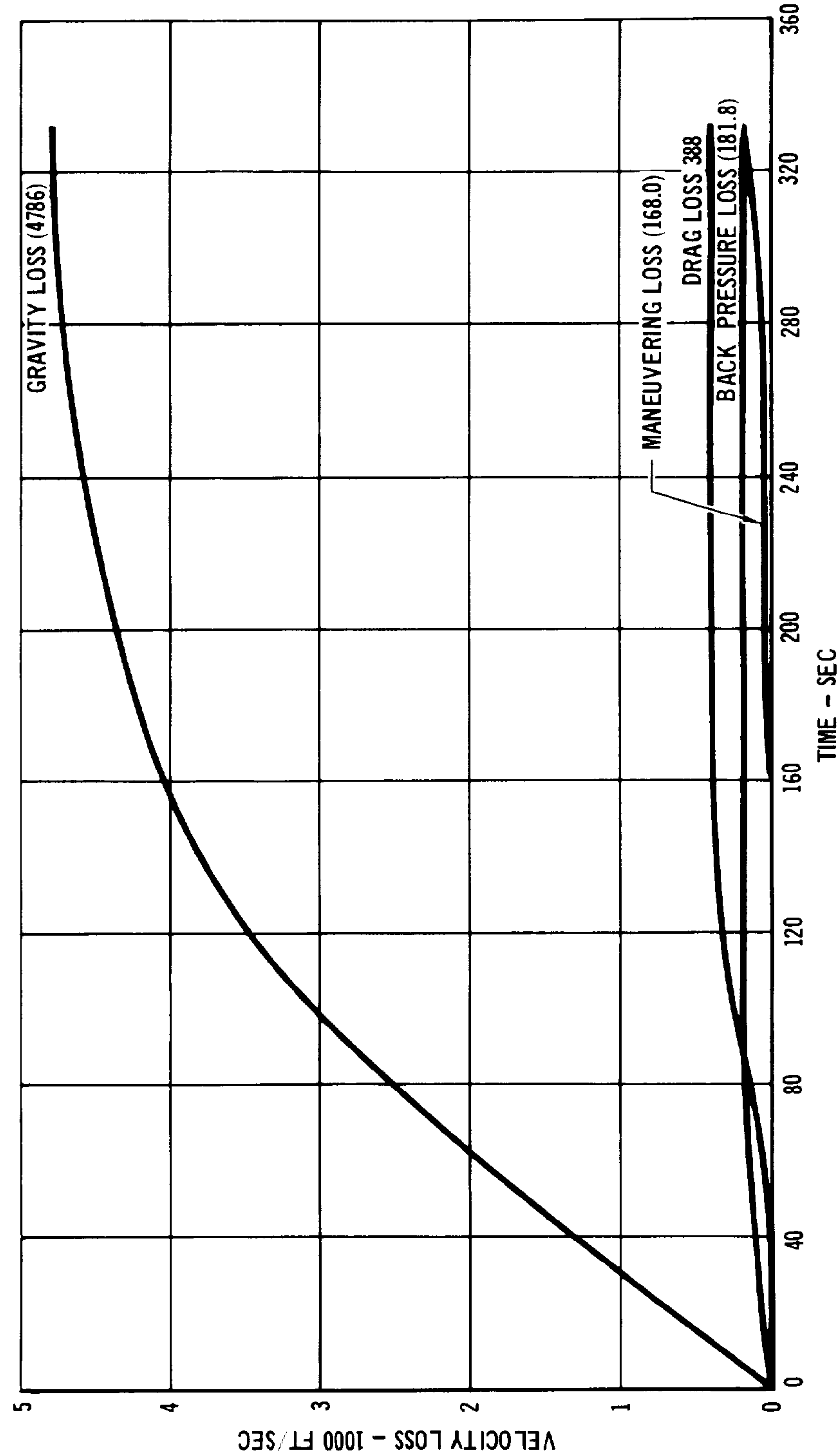


Figure 3-7

performance. A more rational conslusion, however, is that the originial trajectory (4g) was not completely optimized; that the one percent difference is in the tolerance band of these initial studies; and finally, the effect of constraining the maximum acceleration of 3g rather than 4g is a second-order effect. Time hisotries of pertinent trajectory parameters and velocity losses are presented in Figures 3-8, 3-9, and 3-10 for the 3g limit trajectory.

3.2.2 Payload Sensitivity - Deviations in the principal sizing parameters (tank structural fraction, propellant specific impulse, spacecraft subsystems, and the required total ideal velocity) can seriously degrade the performance of a launch vehicle which ultimately requires off-loading of useful cargo in order to maintain acceptable flight performance. A hedge against such an eventuality is obtained by designing a performance margin into the vehicle based on a statistical combination of the payload deviations resulting from these principal sources.

Payload sensitivity to estimated 3 deviations of the principal sizing parameters was evaluated and a design allowance based on a statistical combination (RSS) is presented for each of the stage-and-one-half (4 tip-tanks) designs configured for useful cargo weights of 25,000, 35,000 and 50,000 lbs. respectively. (See Figures 3-11, 3-12, and 3-13.) Nominal mass properties for the three subject configurations were modified for perturbations in structural fraction and payload. These weight statements were used to compute ideal burnout velocity for nominal and 3 perturbation. Sensitivities of payload to structural fraction, propellant I_{sp}, and required velocity are summarized in Table 3-1.

The sensitivity of payload to subsystem weights is one-to-one; each excess pound of subsystem weight requires off-loading one pound of payload. This results from the fact that the rocket velocity is sensitive only to the total injection weight. The 3 variations in the weights of the major subsystems were determined from historical data available form previous studies and development programs.

Aerodynamic surfaces, structure, and thermal protection subsystems weight uncertainties were estimated to be 1.5, and 1.0 pounds per square foot of projected area, respectively. The nominal value was different for each of the 3 designs as shown in Figures 3-14, 3-15 and 3-16, and therefore the percent variation for equivalent subsystems was not a single value. However, the average 3 variations for the structure and aerodynamic surfaces were approximately 26 per-

ILRV LAUNCH TRAJECTORY

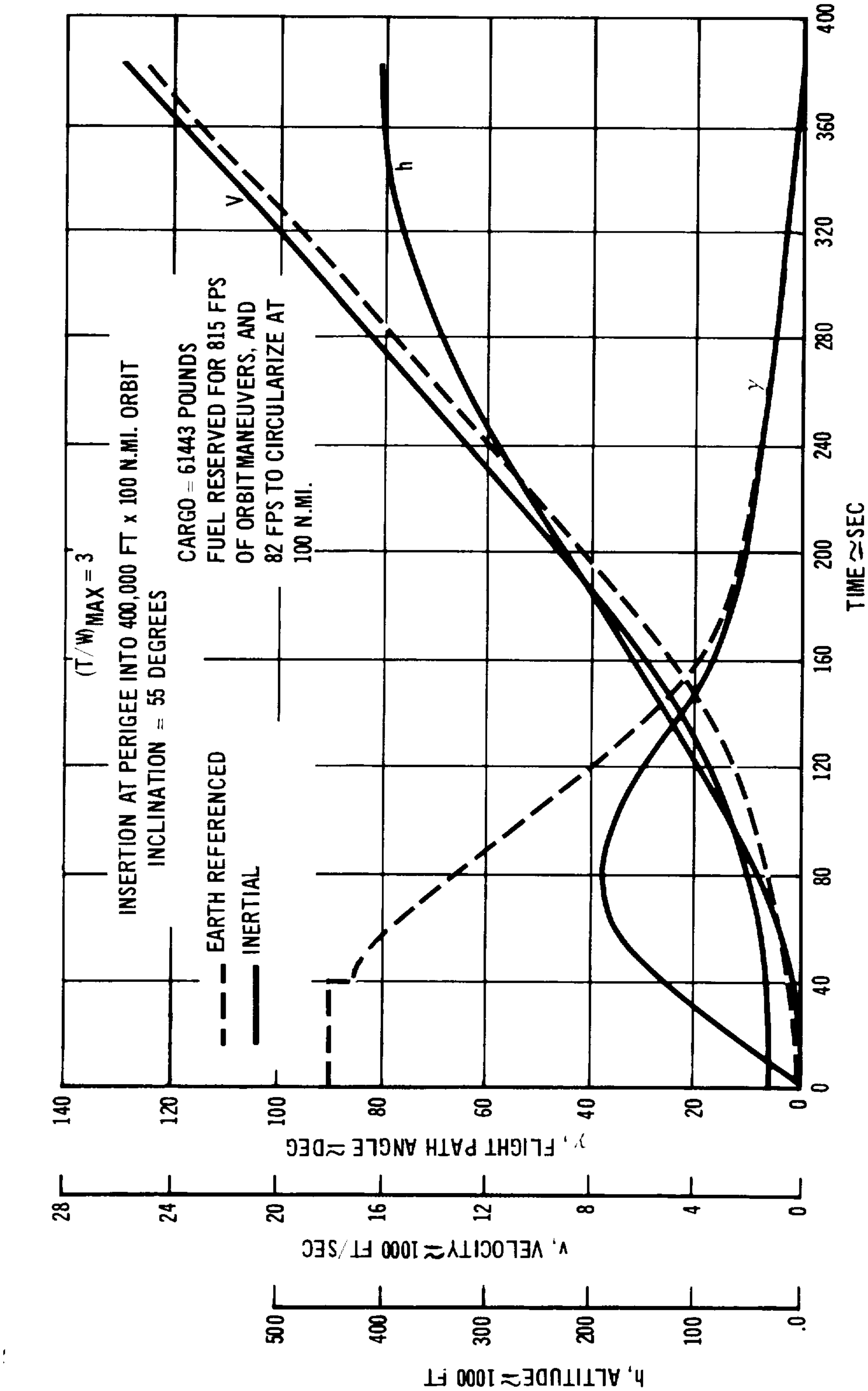


Figure 3-8

ILRV LAUNCH TRAJECTORY

$(T/W)_{MAX} = 3$

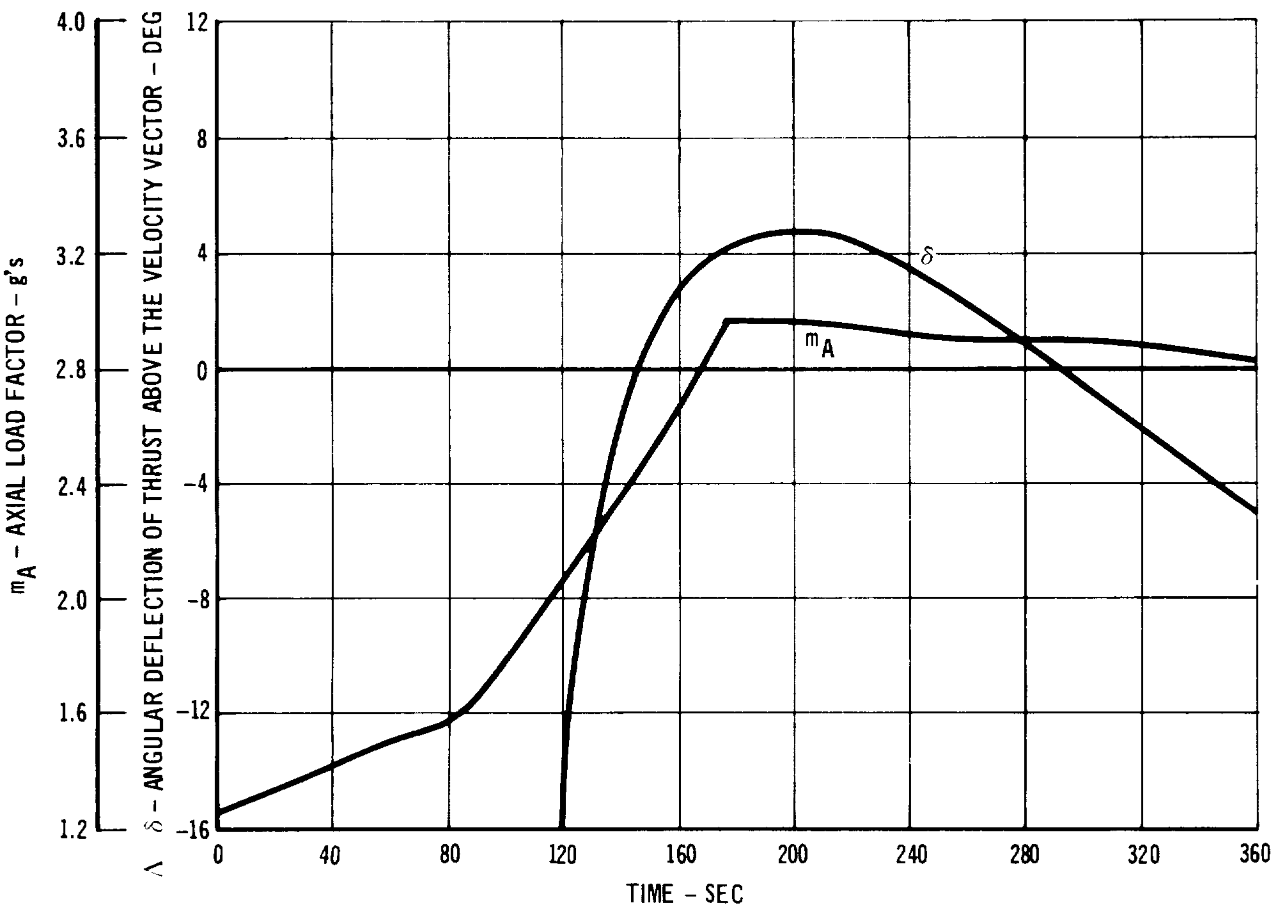


Figure 3-9

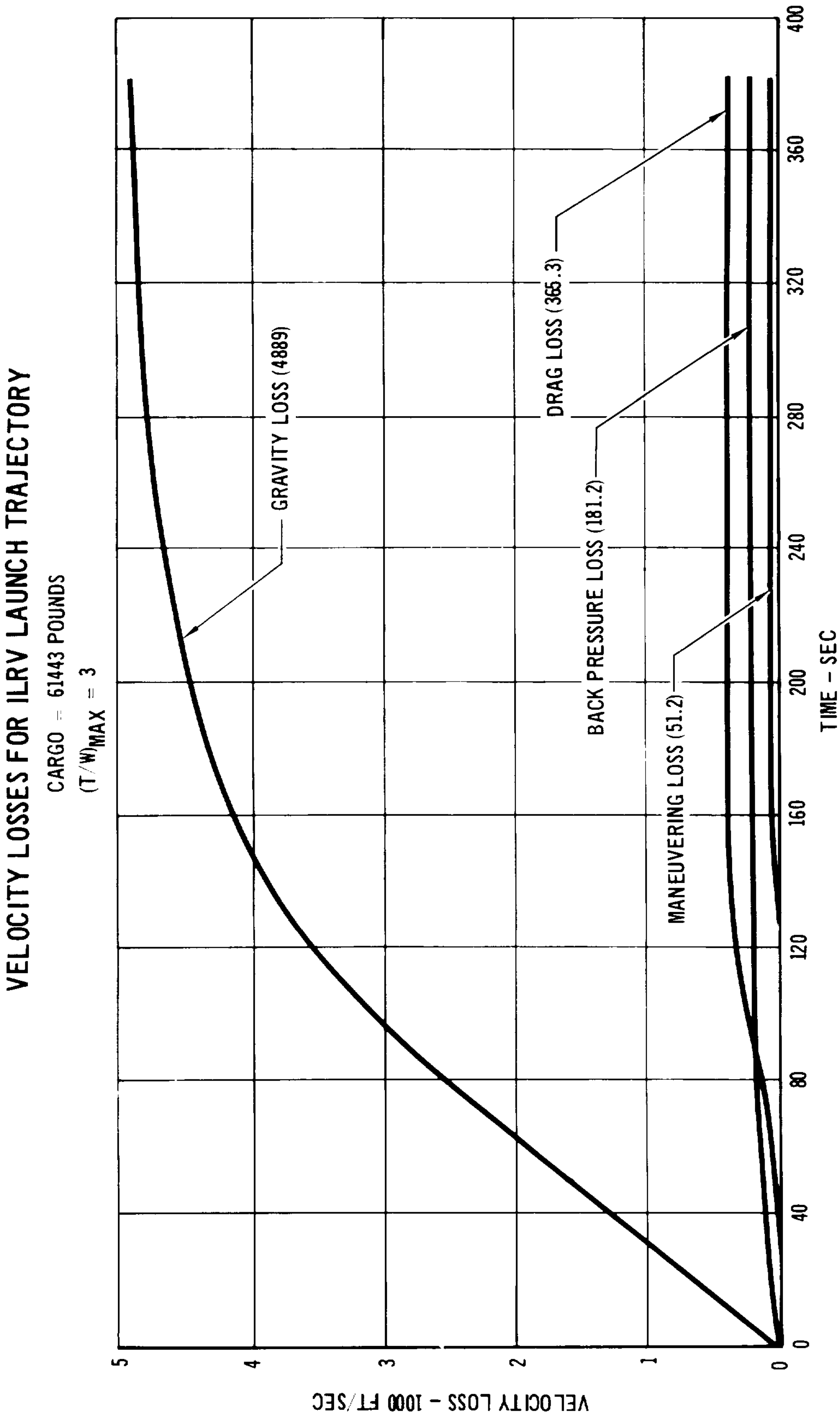

VELOCITY LOSSES FOR ILRV LAUNCH TRAJECTORY
CARGO = 61443 POUNDS
$(T/W)_{MAX} = 3$
GRAVITY LOSS (4889)
DRAG LOSS (365.3)
BACK PRESSURE LOSS (181.2)
MANEUVERING LOSS (51.2)
TIME - SEC
VELOCITY LOSS - 1000 FT/SEC
0
40
80
120
160
200
240
280
320
360
400
0
1
2
3
4
5

Figure 3-10

FOUR TIP-TANK STAGE-AND-ONE-HALF
PAYLOAD SENSITIVITY TO PRINCIPAL SIZING PARAMETERS
25,000 Lb Nominal Payload

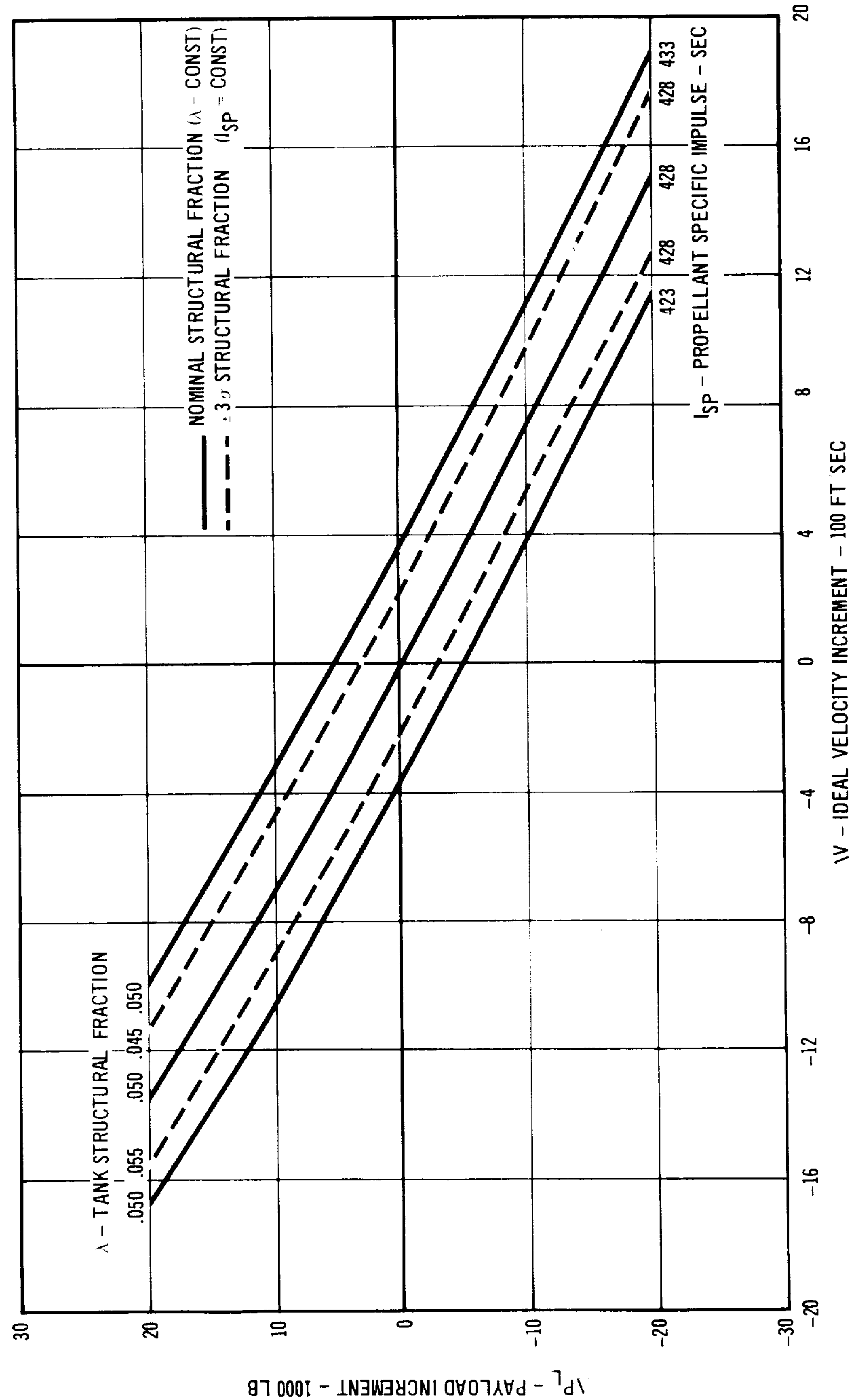


Figure 3-11

FOUR TIP-TANK STAGE-AND-ONE-HALF PAYLOAD SENSITIVITY TO PRINCIPAL SIZING PARAMETERS

35,000 Lb Nominal Payload

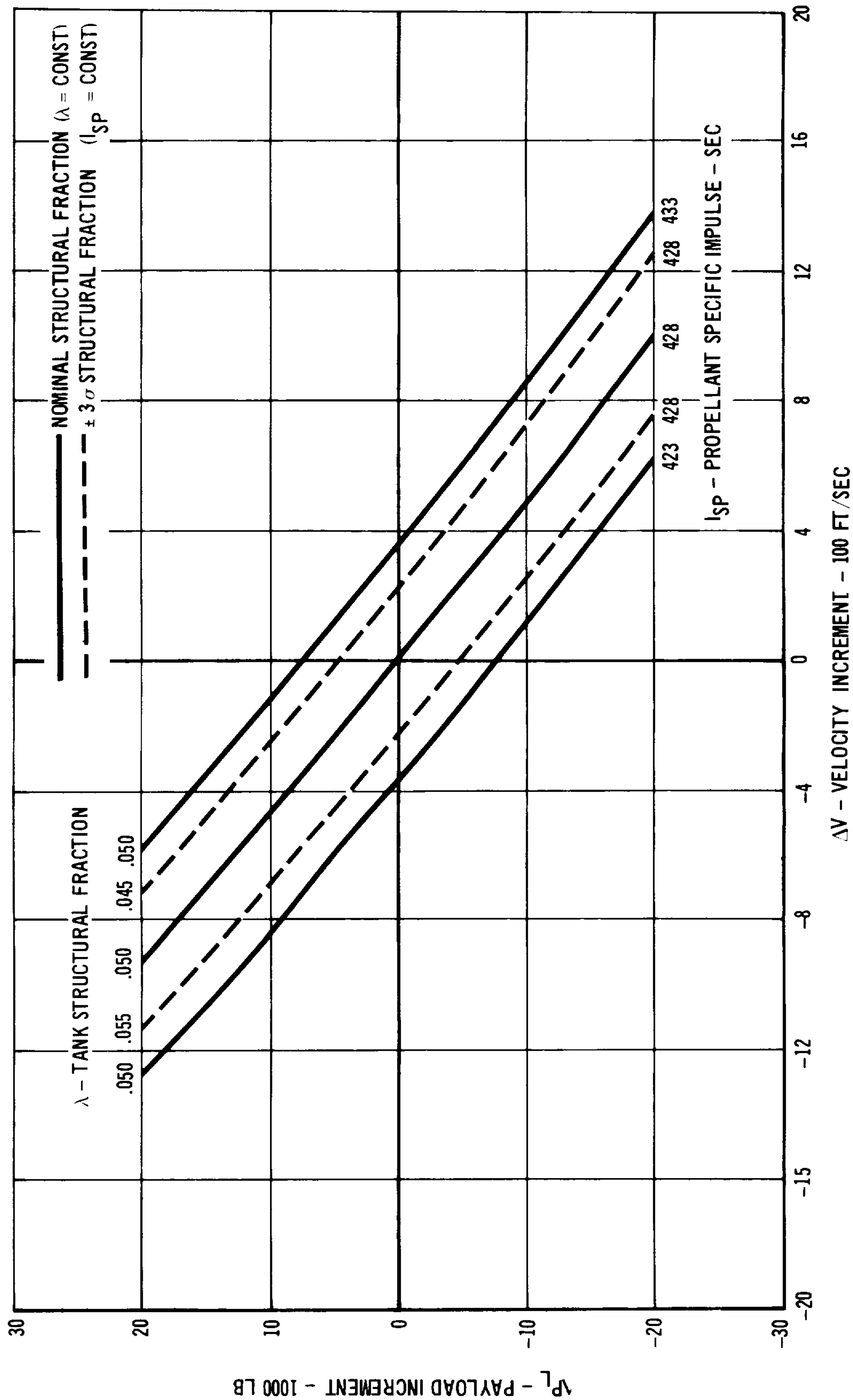


Figure 3–12

FOUR TIP-TANK STAGE-AND-ONE-HALF
PAYLOAD SENSITIVITY TO PRINCIPAL SIZING PARAMETERS
50,000 Lb Nominal Payload

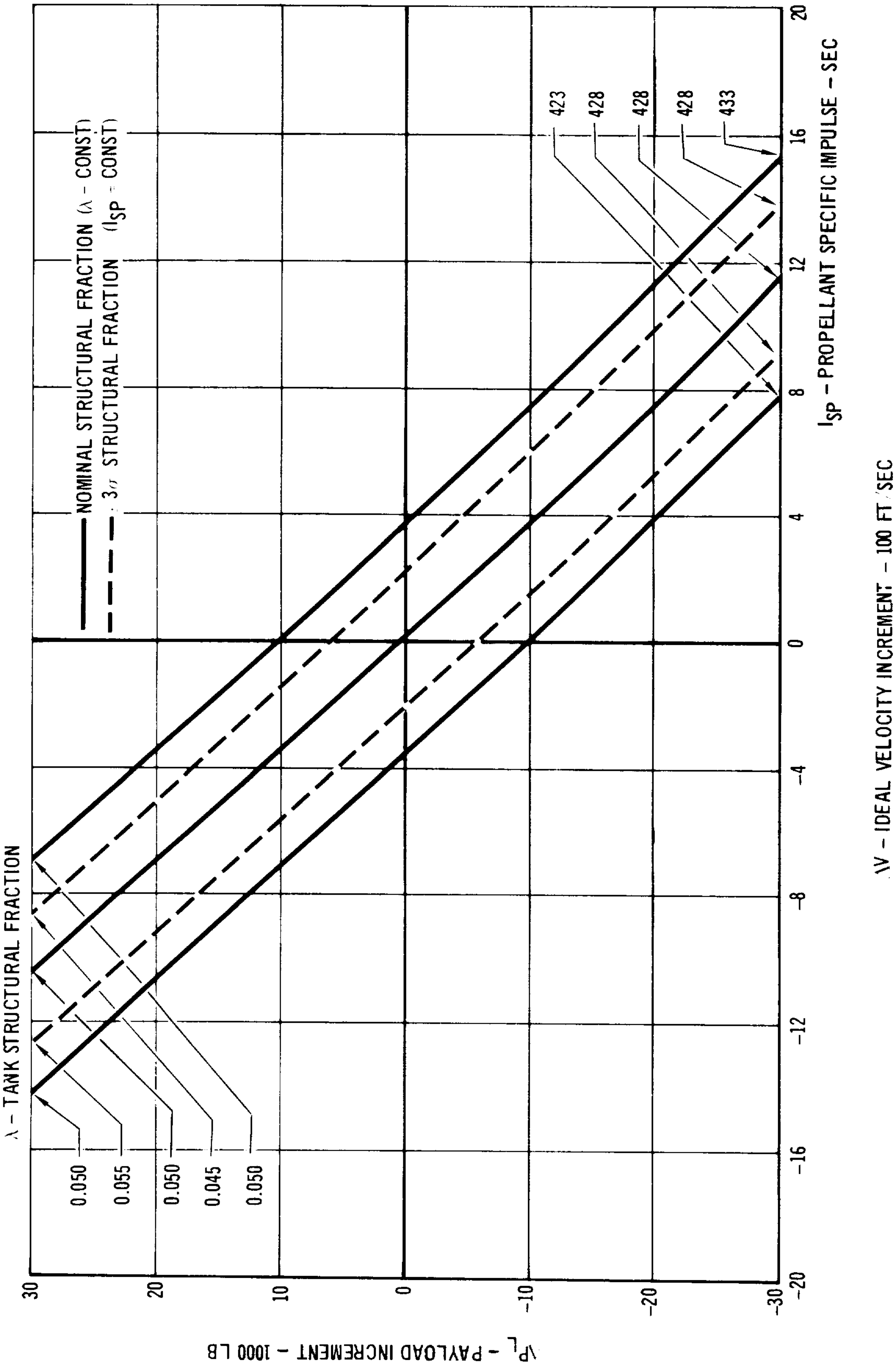


Figure 3-13

Table 3-1

SUMMARY OF PAYLOAD SENSITIVITIES

Useful Cargo Wt. (Lb.)	25,000	35,000	50,000
Landing Wt. (Lb.)	131,000	201,000	250,000
Injection Wt. (Lb.)	146,000	214,370	280,000
Gross Launch Wt. (Lb.)	1,842,000	2,737,170	3,693,200
*Structural Fraction; $\frac{\delta P_L}{\delta \lambda}$ (Pounds Per Percent)	6500	9300	11,400
*Propellant I_{sp}; $\frac{\delta P_L}{\delta I_{sp}}$ (Pounds Per Second)	1050	1550	2,000
*Velocity; $\frac{\delta P_L}{\delta I_{sp}}$ (Pounds Per Ft/Sec)	13.7	20.7	27.3

* Based on average of (+) and (-) values shown in Figures 3-14, 3-15 and 3-16.

FOUR TIP-TANK STAGE-AND-ONE-HALF PAYLOAD EQUIVALENTS FOR 3 σ DEVIATION (25,000 Lb Nominal Payload)

VARIABLE	NOMINAL ± VARIATION	Δ PAYLOAD (LB)	
* * AERO SURFACES	10.9 #/FT^2 ± 3.0 #/FT^2	+2676	−2676
** STRUCTURE	5.64 #/FT^2 ± 1.5 #/FT^2	+5930	−5930
THERMAL PROTECTION	4.17 #/FT^2 ± 1.0 #/FT^2	+3955	−3955
LANDING/RECOVERY	4700 # ± 20%	+ 940	− 940
PRIME POWER	3400 # ± 20%	+ 680	− 680
BOOST ENGINES	27300 # ± 20%	+5450	−5450
BOOST FEED SYSTEM	12300 # ± 20%	+2460	−2460
SUBSYSTEM STATISTICAL COMBINATION (RSS)		+9750	−9750
λ, (STRUCTURAL FRACTION)	.05 ± .005	+3350	−3150
I_{SP}	428 SEC ± 5 SEC	+5350	−5150
V_{TOTAL}	31,186 FPS ± 312 FPS	+4300	−4200
(λ, I_{SP}, V_{TOTAL}) STATISTICAL COMBINATION (RSS)		+7638	−7354
TOTAL STATISTICAL COMBINATION (RSS)		+12390	−12220

* BASED ON PROJECTED AREA
** BASED ON WETTED AREA

Figure 3–14

FOUR TIP TANK STAGE-AND-ONE-HALF PAYLOAD EQUIVALENTS FOR 3σ DEVIATIONS

(35,000 Lb. Nominal Payload)

VARIABLE	NOMINAL ± VARIATION	Λ PAYLOAD (LB)	
*AERO SURFACES	12.8 #/FT^2 ± 3.0 #/FT^2	+3813	-3813
**STRUCTURE	5.75 #/FT^2 ± 1.5#/FT^2	+8490	-8490
**THERMAL PROTECTION	4.36 #/FT^2 ± 1.0#/FT^2	+5660	-5660
LANDING/RECOVERY	6500# ± 20%	+1300	-1300
PRIME POWER	3900# ± 20%	+ 780	- 780
BOOST ENGINES	39000# ± 20%	+7800	-7800
BOOST FEED SYSTEM	19500# ± 20%	+3900	-3900
SUBSYSTEM STATISTICAL COMBINATION (RSS)		+14040	-14040
λ (STRUCTURAL FRACTION)	.05 ± .005	+4800	-4500
I_{SP}	426 SEC ± 5 SEC	+7850	-7600
V_{TOTAL}	31,176 FPS ± 312 FPS	+6500	-6400
(λ, I_{SP}, V_{TOTAL}) STATISTICAL COMBINATION (RSS)		+11,265	-10,907
TOTAL STATISTICAL COMBINATION (RSS)		+18,000	-17,750

* BASED ON PROJECTED AREA
** BASED ON WETTED AREA

Figure 3-15

FOUR TIP TANK STAGE-AND-ONE-HALF PAYLOAD EQUIVALENTS FOR 3σ DEVIATIONS

(50,000 Lb Nominal Payload)

VARIABLE	NOMINAL ± VARIATION	Δ PAYLOAD (LB)	
*AERO SURFACES	11.0 #/FT^2 ± 3.0 #/FT^2	+ 5025	− 5025
** STRUCTURE	5.9 #/FT^2 ± 1.5 #/FT^2	+11120	−11120
** THERMAL PROTECTION	4.18 #/FT ± 1.0 #/FT^2	+ 7417	− 7417
LANDING RECOVERY	7700 # ± 20%	+ 1540	− 1540
PRIME POWER	3900 # ± 20%	+ 780	− 780
BOOST ENGINES	53500 # ± 20%	+10600	−10600
BOOST FEED SYSTEM	25500 # ± 20%	+ 5100	− 5100
SUBSYSTEM STATISTICAL COMBINATION (RSS)		+18580	−18580
λ (STRUCTURAL FRACTION)	.05 ± .005	+ 5700	− 5700
I_{SP}	428 SEC ± 5 SEC	+10250	− 9750
V_{TOTAL}	41,166 FPS ± 317 FPS	+ 8700	− 8600
(λ, I_{SP}, V_{TOTAL}) STATISTICAL COMBINATION (RSS)		+14,603	−14,195
TOTAL STATISTICAL COMBINATION (RSS)		+23,600	−23,390

* BASED ON PROJECTED AREA
** BASED ON WETTED AREA

Figure 3-16

cent of currently estimated weights. The thermal protection system averaged 24 percent. All other subsystems were varied by 20 percent.

The payload equivalent of 3 deviations in major subsystem weights is summarized in tabular form in Figures 3-14, 3-15, and 3-16. Also included are the payload equivalents of 3 deviations in structural fraction, propellant I_{sp}, and required total velocity. The cumulative effect on payload of these deviations was determined by root-sum-square (RSS). The total statistical combination for each design represents the possible payload penalty which may be experienced as a result of design uncertainites. Alternately, these values may be used to define an overweight payload for vehicle sizing purposes to insure satisfactory flight performance even in the presence of adverse deviations.

The effect on payload capability due to 3 deviations in subsystem weight is approximately 30 percent greater than the effect due to the sizing parameters , I_{sp}, and V_{Total}. The total statistical combination of all 3 variations is equal to about 50% of the nominal payload for each of the 25,000, 35,000, and 50,000 lb. cases.

3.2.3 Reentry and Glide - An investigation of reentry and glide was made to determine the effects of W/S and C_L on the parameters, altitude, velocity and angle of attack. Trajectories at maximum C_L and maximum L/D were computed for vehicles with a W/S of 60 and 120 pounds per square foot. Figure 3-17 presents the resulting altitude-velocity profiles.

3.3 Total Impulsive Velocity Requirements - Total impulsive velocity requirements, shown in Figure 3-18, for 100, 270, and 450 na mi target altitudes, include the impulsive velocity for the ascent phase plus the transfer requirements. The ascent velocity requirements included in the total are for injection into a 100 na mi circular orbit for inclinations from 28 degrees to 145 degrees which are the limiting inclinations attainable from the CONUS without yaw steering and/or orbital plane change.

Ascent impulsive velocity requirements range between 30,500 ft/sec to 33,000 ft/sec for orbit inclinations attainable from the CONUS. The principal parameters which define the gross launch weight are the payload weight and the total velocity which must be provided by the boost propulsion system. For a non-rotating Earth assumption the total velocity consists of the actual injection velocity plus the velocity increments lost during the ascent phase to overcome

EFFECTS OF W/S AND C_L ON REENTRY PROFILES

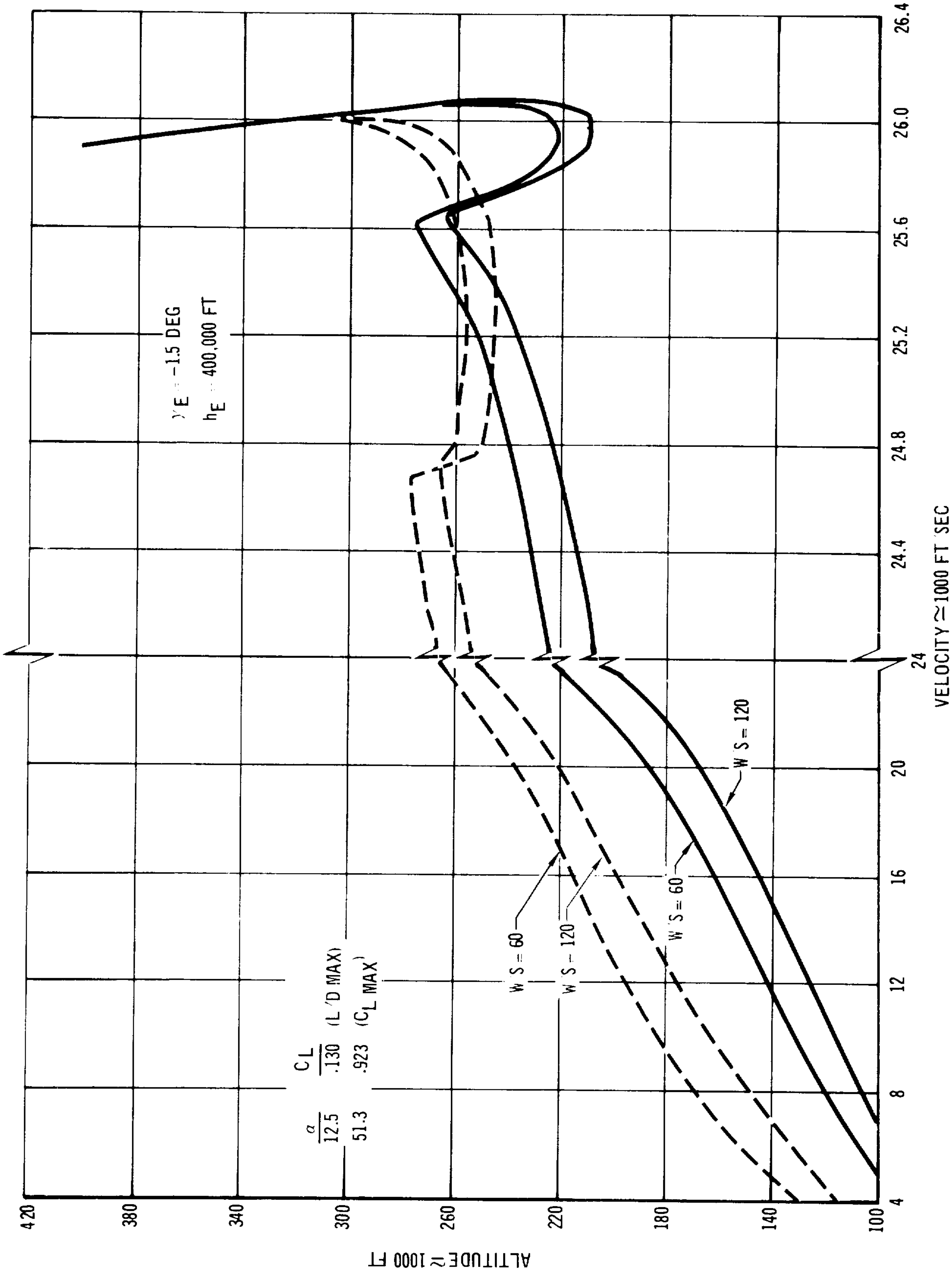
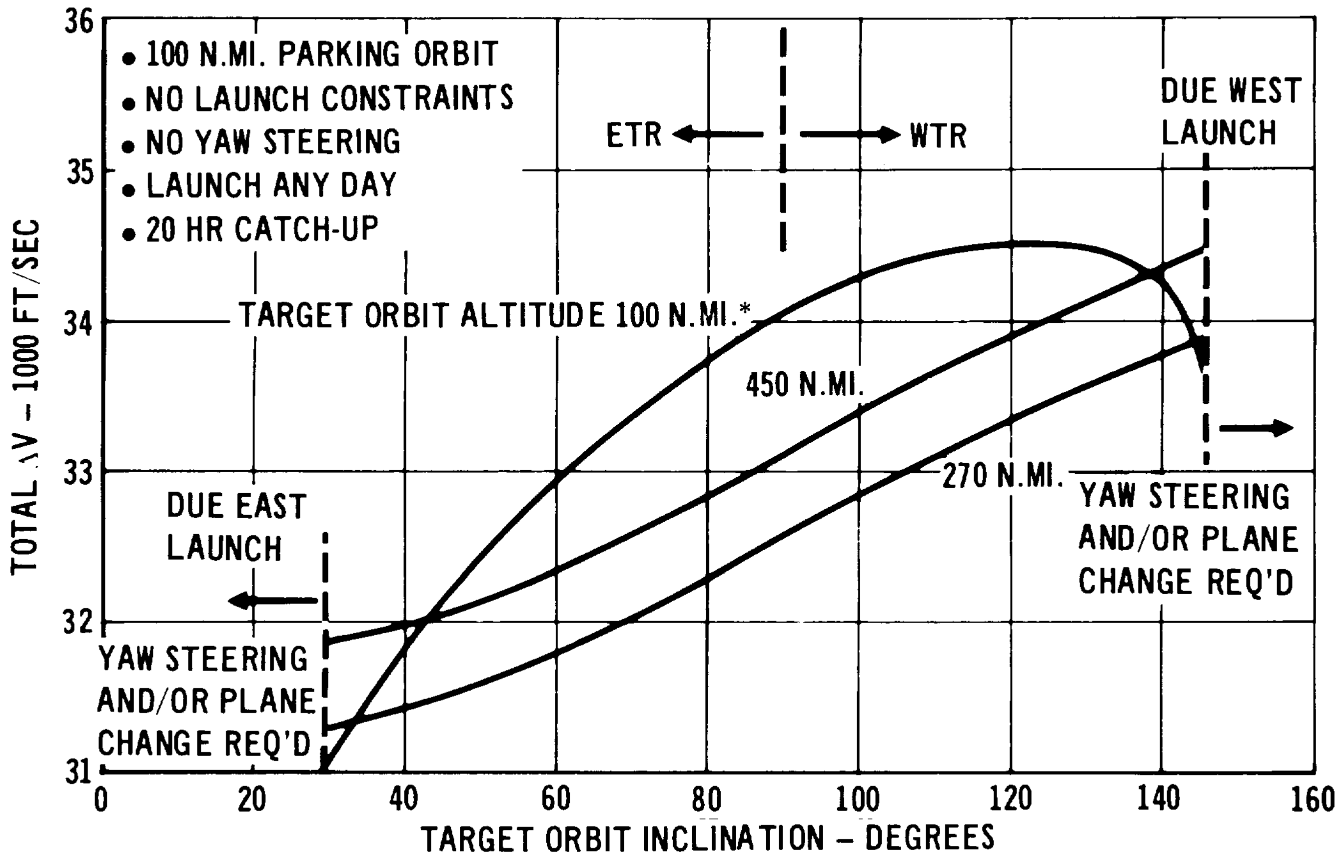


Figure 3-17

TOTAL IMPULSIVE VELOCITY REQUIREMENTS



*100 N.MI. CURVE INCLUDES PLANE CHANGE.

Figure 3–18

gravity, drag, etc. Earth rotation reduced the requirement for easterly launches and increases the requirement for westerly launches. In both cases the increment is a function of launch azimuth.

The shape of the western coastline of CONUS allows a wider launch azimuth sector for WTR compared with the launch sector of ETR. Modified launch azimuth constraints would permit inclinations from 28 degrees to 57 degrees from ETR and inclinations from 62 degrees to 145 degrees from WTR. Thus, the modified constraints eliminate only a small number or orbits; i.e., inclinations from 57 to 62 degrees. Yaw steering during boost could be used to retain these 5 degrees of inclination in the spectrum of attainable orbits from CONUS.

Total impulsive velocity requirements are shown in Figure 3-18 as a function of target orbit altitude and inclination for 100, 270, and 450 na mi altitudes from a 100 na mi parking orbit. No launch azimuth constraints were included, but a 24 hour ascent requirement was imposed which allows a maximum of 20 hours in the parking orbit.

The higher altitude orbits easily meet the ascent requirement without plane change. The 100 na mi target altitude requires plane changes approaching 5.0 degrees for 90 degrees inclination. The rapidly increasing impulsive velocity requirements reflect this plane change requirement.

3.3.1 <u>Payload Capability vs Mission Requirement</u> - Payload capability for various mission requirements for the 130 foot vehicle with 50,000 lb payload is shown in Figure 3-19. At 270 na mi and 55° orbit inclination a 50,000 lb payload requires about 31,700 ft/sec of impulsive velocity. To carry the same payload to 450 na mi and 55° the vehicle must provide about 32,300 ft/sec and 32,750 ft/sec to carry the same payload to a target in the 100 na mi orbit at 55° inclination. For increased target orbit inclinations, the impulsive velocity increases. For parametric calculations, with increasing impulsive velocity requirements, cargo was removed and propellant volume was allocated at 85% of the replaced cargo volume. For decreasing impulsive velocity requirements, cargo density was varied, with constant volume, to provide increased cargo weight capability.

PAYLOAD CAPABILITY VS. MISSION REQUIREMENTS

- MODEL 176M WITH 4 TIP TANKS
- PAYLOAD = 15' x 60' CANISTER
- ORBIT INCLINATION = 90^0

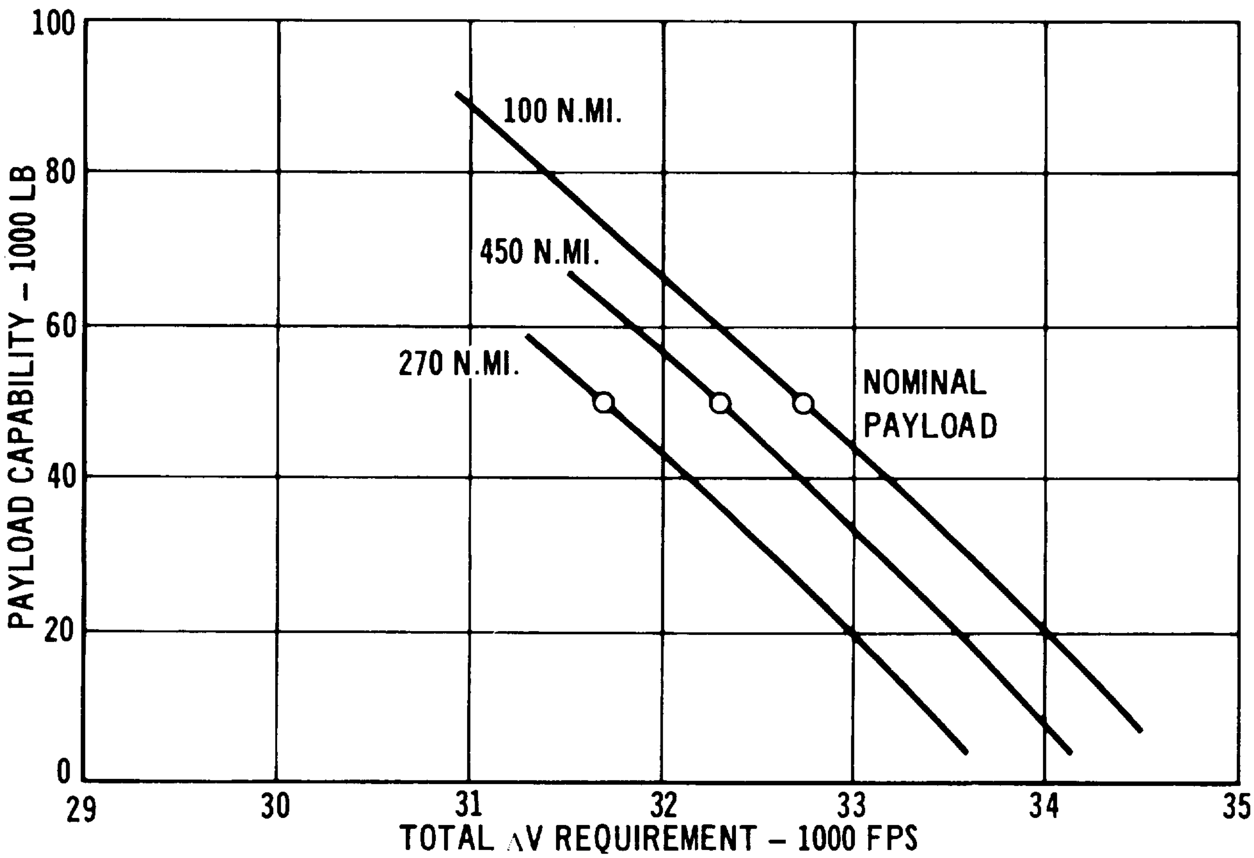


Figure 3–19

Figure 3-20 shows the case where the nominal paylaod of 50,000 lb. is injected into a 100 na mi (mission altitude) 90° inclination orbit. There is no orbital phasing included. Thus, for the 50,000 lb. nominal payload, impulsive velocity requirements for the 90° case, without phasing, are about 1000 ft/sec less than the 55° case, with phasing, shown in Figure 3-18.

3.3.2 Payload Capability - In the payload capability versus orbital inclination plot, Figure 3-21, the curves cross at the nominal mission point of 50,000 lbs. payload and 55° inclination. As inclination increases, the impulsive velocity requirements for the 100 na mi case increase rapidly due to plane changes, so that the payload capability decreases below that for the 270 and 450 na mi cases. Total impulsive velocity requirements for the 100 na mi case reach a maximum at about 122° inclination, so payload capability reaches a minimum at this inclination. The 100 na mi curve then crosses the 270, 450 na mi curve and begins to increase.

The payload capability plot for a 100 na mi mission altitude and a nomina; orbital inclination of 90° is shown in Figure 3-22. The shape of the curve is identical to the 100 na mi, 55° inclination curve shown in Figure 3-21 except it is shifted upward. Elimination of orbital phasing with its attendant impulsive velocity requirements, shifts the payload weight upward approximately 18,000 lb. for the same orbital inclination.

PAYLOAD CAPABILITY VS. MISSION REQUIREMENTS

- MODEL 176M WITH 4 TIP TANKS
- 15' X 60' PAYLOAD CANISTER
- MISSION ALTITUDE = 100 N.MI.
- NO ORBITAL PHASING
- ORBIT INCLINATION = 90^0

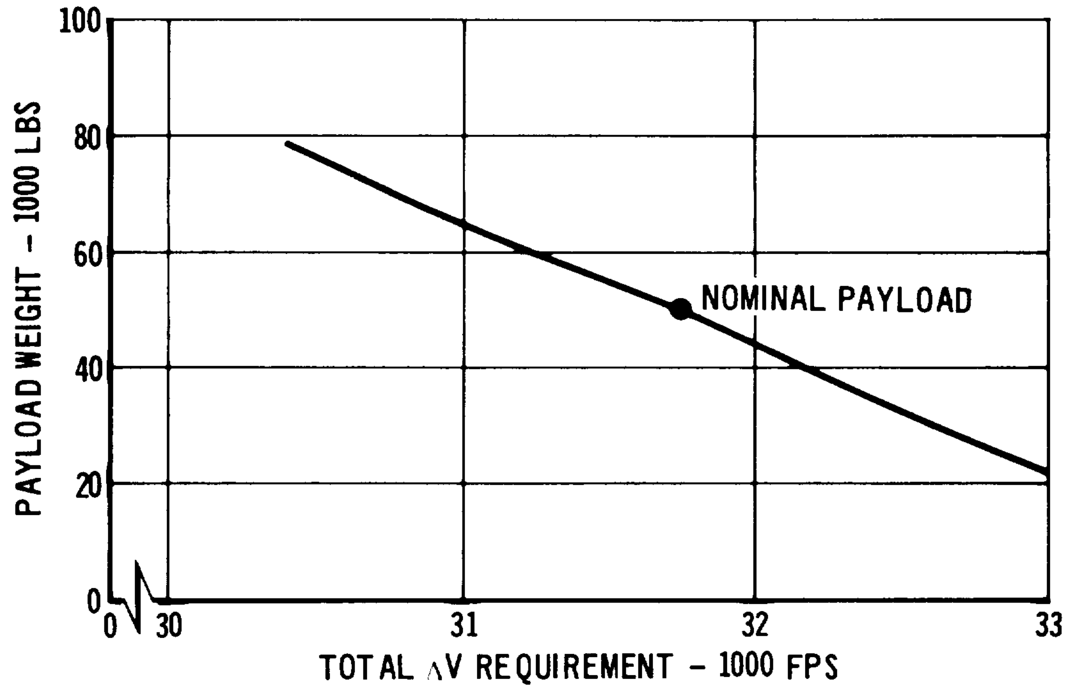


Figure 3-20

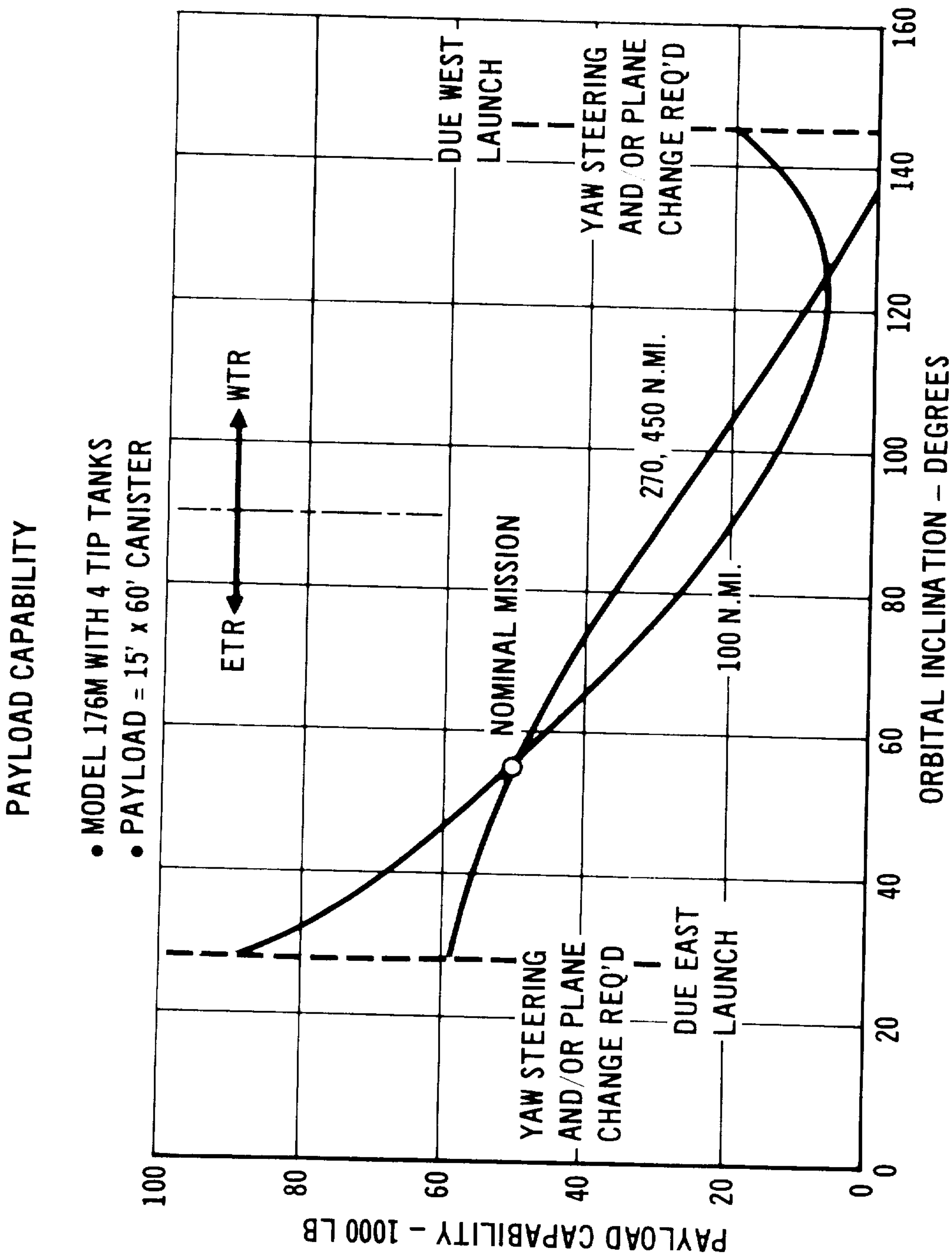

PAYLOAD CAPABILITY
• MODEL 176M WITH 4 TIP TANKS
• PAYLOAD = 15' x 60' CANISTER
ETR
WTR
NOMINAL MISSION
DUE WEST LAUNCH
YAW STEERING AND/OR PLANE CHANGE REQ'D
270, 450 N.MI.
100 N.MI.
YAW STEERING AND/OR PLANE CHANGE REQ'D
DUE EAST LAUNCH
PAYLOAD CAPABILITY – 1000 LB
0
20
40
60
80
100
ORBITAL INCLINATION – DEGREES
0
20
40
60
80
100
120
140
160

Figure 3-21

PAYLOAD CAPABILITY

- MODEL 176M WITH 4 TIP TANKS
- 15' X 60' PAYLOAD CANISTER
- MISSION ALTITUDE ≈ 100 N.MI.

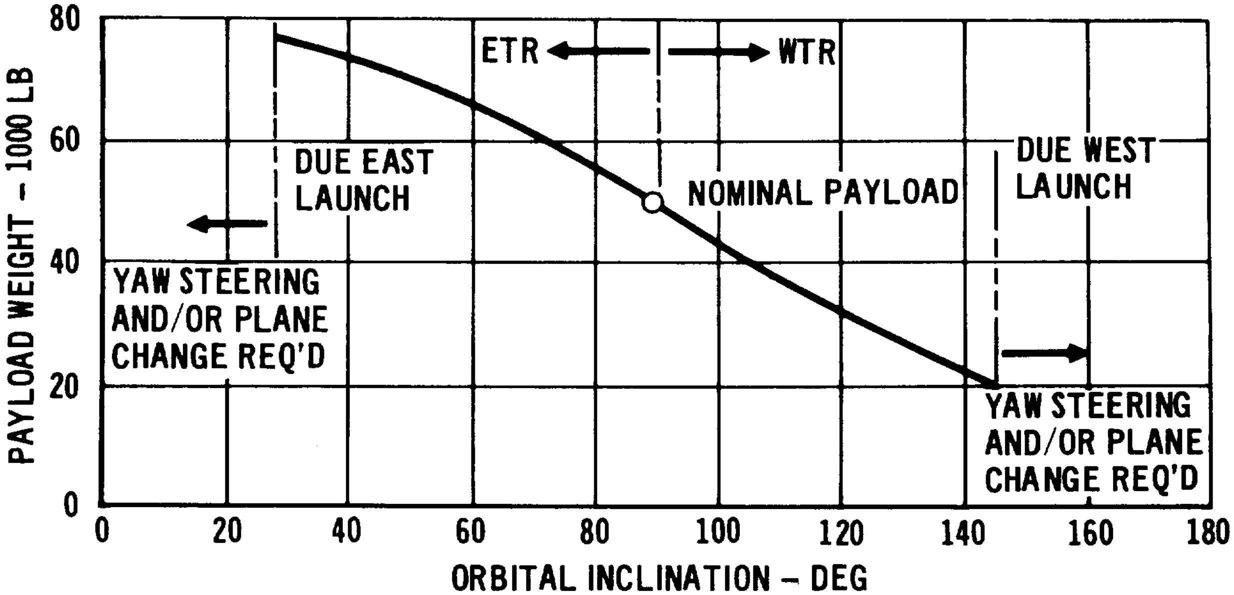


Figure 3-22

4.0 OPERATIONAL MODES

In this section, the mission profile and sequence of events are presented for the baseline 1-1/2-stage vehicle. In addition, specific operational modes such as payload-core vehicle integration, payload-payload canister integration, the swing-nose concept, and alternate mission modes have been investigated and are discussed here.

4.1 Mission Profile - A pictorial major-event sequence for the baseline mission is shown in Figure 4-1. The logistic mission consists of resupplying men, food, equipment, tools, experiments, etc. to a space station in a 55-degree-inclined, 270-NM-circular Earth orbit. The space vehicle is launched from the Eastern Test Range (ETR) along a 139-degree azimuth. During ascent, the external tanks are jettisoned, serially and in pairs (side tanks first). The core vehicle then uses internal propellant to inject itself into a 45 x 100 NM parking orbit, which is later circularized to 100 NM. The vehicle then coasts in the parking orbit until proper phasing with the space station occurs, whereupon it transfers to the 270-NM space-station altitude where rendezvous and payload transfer is accomplished. After a nominal 5-day stay in orbit, the core vehicle, carrying a return payload, returns to Earth. The primary landing site is located near the launch site. Upon landing, the vehicle is recycled through a recertification phase and moved to the launch pad in preparation for the next flight.

4.2 Mission Sequence of Events - A typical detailed sequence of events for the occurrences in a space station logistics mission is given in Table 4-1. Here, the mission is divided into five mission phases; namely, prelaunch operations, ascent, orbital operations, descent, and maintenance operations. Major events, event-initiation times and event-duration times are given under each phase.

ILRV LOGISTICS MISSION PROFILE

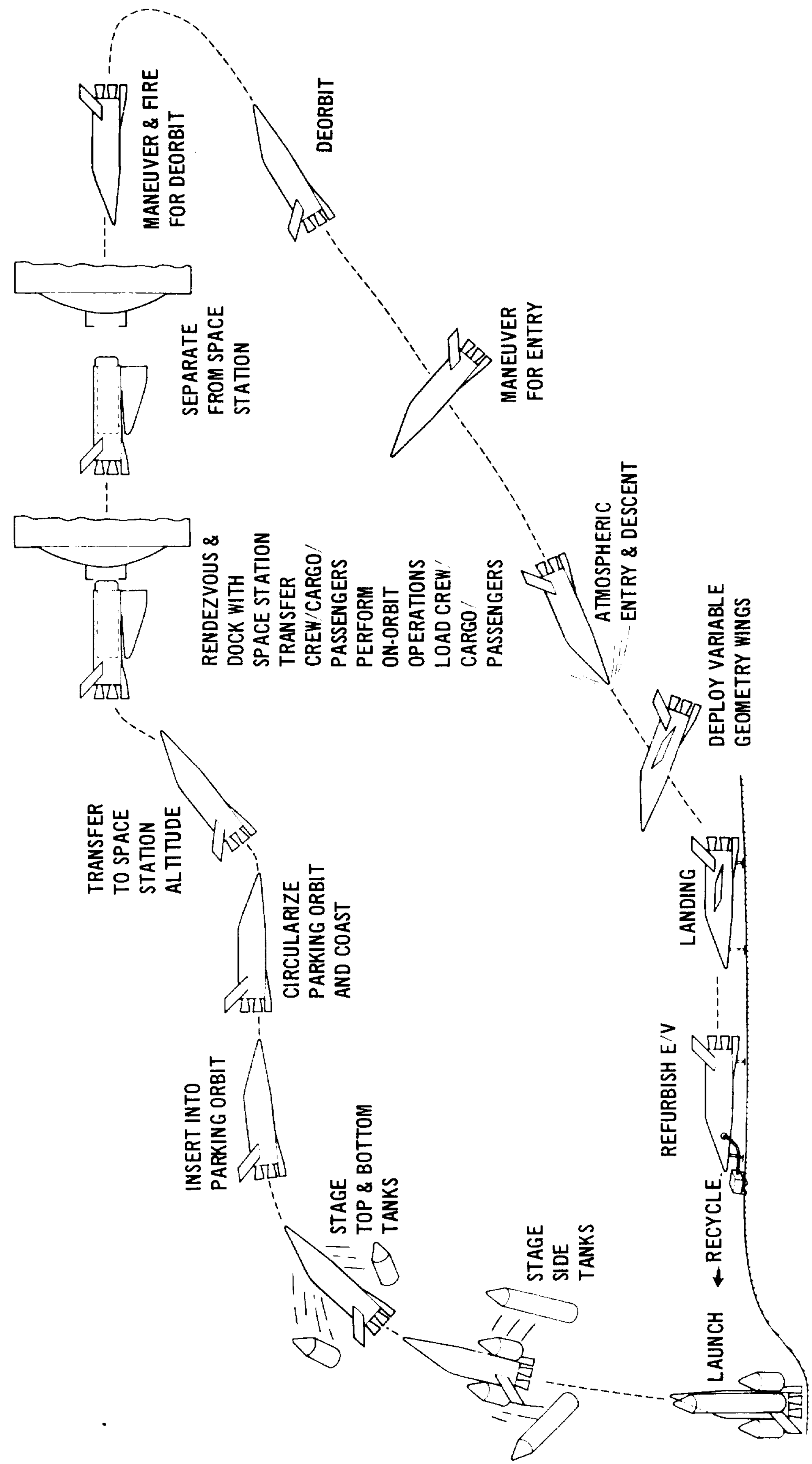


Figure 4-1

Table 4-1 (1)

SEQUENCE OF EVENTS - LOGISTICS MISSION

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
0.0 PRELAUNCH OPERATIONS PHASE			
0.1 Transport Core Vehicle to Launch Pad	T_o - 24 hrs.	1 hr.	
0.2 Erect Core Vehicle on Pad	T_o - 23 hrs.	5 hrs.	
0.3 Transport Drop Tanks to Pad	T_o - 18 hrs.	1 hr.	
0.4 Erect Drop Tanks	T_o - 17 hrs.	4 hrs.	
0.5 Mate Drop Tanks to Core Vehicle	T_o - 13 hrs.	3 hrs.	
0.6 Connect Holddowns	T_o - 10.5 hrs.	30 min.	
0.7 Hookup Cryogenic Service Lines	T_o - 10 hrs.	2 hrs.	
0.8 Perform Tank Leakage Test	T_o - 8 hrs.	2 hrs.	
0.9 Power up for Range Check and Navigational Input	T_o - 6 hrs.	2 hrs.	
0.10 Propulsion Subsystem Operational Checkout	T_o - 6 hrs.	2 hrs.	
0.11 Final Preparation and Inspection	T_o - 4 hrs.	1 hr.	
0.12 Crew Ingress	T_o - 3.5 hrs.	20 min.	
0.13 Final Systems Checkout and Guidance Update	T_o - 3 hrs.	1 hr.	
0.14 Crew Egress	T_o - 2 hrs.	15 min.	Safety measure.

Table 4-1(2)

SEQUENCE OF EVENTS - LOGISTICS MISSION

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
0.15 Begin Cryogenic Servicing	T_o - 2 hrs.	1.5 hrs.	
0.16 Begin LH_2 Precool	T_o - 120 min.	15 min.	
0.17 Begin LH_2 Slow Fill	T_o - 105 min.	18 min.	to 5%.
0.18 Begin LH_2 Fast Fill	T_o - 87 min.	39 min.	to 95%.
0.19 Begin LO_2 Precool	T_o - 81 min.	15 min.	
0.20 Begin LO_2 Slow Fill	T_o - 66 min.	6.5 min.	to 5%
0.21 Begin LO_2 Fast Fill	T_o - 59.5 min.	24.5 min.	to 96.5%
0.22 Begin LH_2 Slow Fill/Topping	T_o - 48 min.	18 min.	to 100%
0.23 Begin LO_2 Slow Fill/Topping	T_o - 35 min.	5 min.	to 100%
0.24 Crew Ingress	T_o - 35 min.	20 min.	
0.25 Passenger Ingress	T_o - 30 min.	20 min.	
0.26 Boarding GSE Removal	T_o - 10 min.	2 min.	
0.27 Final Countdown	T_o - 10 min.	10 min.	

Table 4-1 (3)

SEQUENCE OF OPERATIONS - LOGISTICS MISSION

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
1.0 ASCENT PHASE			
1.1 Ignite Engines	T_o - 3 sec.	-	
1.2 Holddown Space Vehicle	T_o - 3 sec.	3 sec.	
1.3 Release Holddowns (Lift-off)	T_o	-	
1.4 Begin Gravity Turn	T_o + 20 sec.	100 sec.	
1.5 Translate Engine Nozzles	T_o + 65 sec.	-	
1.6 Experience Max.-g	T_o + 72 sec.	-	h = 34,000 ft. q_{MAX} = 500 psf
1.7 Jettison Side Drop Tanks	T_o + 148 sec.	-	h = 95,000 ft.; r = 35 NM; ΔV_1 = 9350 fps (ideal)
1.8 Begin Constant-4g Acceleration	T_o + 192 sec.	120 sec.	
1.9 Jettison Upper and Low Drop Tanks	T_s	-	$T_s = T_o + 226$ sec.; ΔV = 9350 fps (ideal); h = 200,000 feet; r = 150 NM.
1.10 Burn Core Vehicle into Orbit	T_s	100 sec.	ΔV_3 = 12,200 fps (ideal); 45 x 100-NM parking orbit; r = 450 NM.
1.11 Coast in Parking Orbit	T_s + 100 sec.	0-88 min.	

Table 4 -1 [4]

SEQUENCE OF OPERATIONS - LOGISTICS MISSION

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
1.12 Circularize Parking Orbit	T_s + 48.8 min.	-	Nominal time; circularize to 100 NM; ΔV = 201 fps.
1.13 Coast in Parking Orbit	T_s + 49 min.	0-20 hrs.	Phasing with Space Station.
1.14 Transfer to 255-NM Orbit	T_s + 20 hrs.	96 min.	3-impulse maneuver; ΔV = 630 fps; S/C approx. 15 NM below and 75 NM behind Station; max. time given.
1.15 Coast in 255-NM Orbit	T_s + 21.5 hrs.	30 min.	
1.16 Transfer to 270-NM Orbit	T_s + 22 hrs.	66 min.	2-impulse maneuver; ΔV = 200 fps; S/C approx. 3 NM behind Station.

Table 4-1 (5)

SEQUENCE OF EVENTS - LOGISTICS MISSION

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
2.0 ORBITAL OPERATIONS PHASE			
2.1 Swing Core Vehicle Nose Away	T_x	10 min.	$T_x = T_s + 24$ hrs. (max.)
2.2 Translate Payload Canister Forward	T_x + 10 min.	20 min.	
2.3 Dock with Space Station	T_x + 30 min	30 min.	$\Delta V = 15$ fps
2.4 Withdraw from Payload	T_x + 1 hr.	20 min.	
2.5 Rendezvous with Return Payload	T_x + 80 min.	40 min.	$\Delta V = 10$ fps
2.6 Dock around Return Payload	T_x + 2 hrs.	30 min.	$\Delta V = 15$ fps
2.7 Perform Space Station Support	T_x + 2.5 hrs.	0-5 days	
2.8 Undock from Space Station	T_y		$T_y = T_x + 2.5$ hrs. (minimum time); $T_y = T_x + 5$ days (maximum time)
2.9 Translate Payload Canister to Rear of Core Vehicle	T_y + 10 min.	20 min.	
2.10 Close Core Vehicle Nose	T_y + 30 min.	10 min.	

Table 4-1 (6)

SEQUENCE OF EVENTS - LOGISTICS MISSION

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
3.0 DESCENT PHASE			
3.1 Maneuver for Deorbit	T_Y + 4 hrs.	30 min.	
3.2 Fire Deorbit Motor	T_D		ΔV = 400 fps. $T_D = T_Y + 4.5$ hrs.
3.3 Burn Deorbit Motor	T_D	10 sec.	
3.4 Terminate Deorbit Burn	T_D + 10 sec.	-	
3.5 Descend from Orbit	T_D + 10 sec.	32 min.	
3.6 Maneuver for Entry	T_D + 60 sec.	10 min.	
3.7 Enter Atmosphere	T_E	-	h = 400,000 $T_E = T_D + 32$ min.
3.8 Descend Through Atmosphere	T_E	3670 sec.	
3.9 Extend Variable-Geometry Wings	T_E + 3500 sec.	-	h = 30,000 ft.
3.10 Deploy Go-Around Engines	T_E + 3510 sec.	-	
3.11 Start Go-Around Engines	T_E + 3520 sec.	-	
3.12 Burn Go-Around Engines	T_E + 3520 sec.	150 sec.	
3.13 Extend Landing Gear	T_E + 3590 sec.	-	h = 1000 ft.
3.14 Land	T_L	-	$T_L = T_E + 3670$ sec.

Table 4-1 (7)

SEQUENCE OF EVENTS - LOGISTICS MISSION

MISSION EVENT		EVENT INITIATION TIME	EVENT DURATION	REMARKS
4.0	MAINTENANCE PHASE			
4.1	Passenger Egress	T_L + 10 min.	20 min.	Passengers fly to predesignated areas via aircraft.
4.2	Crew Egress	T_L + 10 min.	20 min.	
4.3	Data Removal	T_L + 30 min.	10 min.	Critical data only.
4.4	Install Safety Devices	T_L + 30 min.	12 min.	All propulsive and pyrotechnic systems safed.
4.5	Make Visual Inspection	T_L + 42 min.	15 min.	
4.6	Perform Cabin Switch Check	T_L + 48 min.	15 min.	
4.7	Cool and Decontaminate S/C	T_L + 63 min.	3 hrs.	
4.8	Move S/C to Work Area	T_L + 4 hrs.	30 min.	
4.9	Unload Payload Canister	T_L + 4.5 hrs.	2 hrs.	Payload canister shipped to launch site via Super Guppy aircraft.
4.10*	Load Jet Fuel	T_L + 6.5 hrs.	2 hrs.	
4.11*	Install Strap-On Ferry Engines	T_L + 8.5 hrs.	6.5 hrs.	
4.12*	Crew Ingress	T_L + 15 hrs.	20 min.	
4.13*	S/C Horizontal Takeoff	T_L + 16 hrs.	-	
4.14*	Fly to Launch Site	T_L + 16 hrs.	0-8 hrs.	
4.15*	Land at Launch Site	T_B	-	Maximum time. $T_B = T_L + 24$ hrs.

* Applies to landing at remote site only.

Table 4-1[8]

SEQUENCE OF EVENTS - LOGISTICS MISSION

	MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
4.16	Crew Egress	T_B + 10 min.	20 min.	
4.17	Install Safety Devices	T_B + 30 min.	12 min.	All propulsive and pyrotechnic systems safed.
4.18	Make Visual Inspection	T_B + 42 min.	15 min.	
4.19	Perform Cabin Switch Check	T_B + 48 min.	15 min.	
4.20	Move S/C to Maintenance Area	T_B + 1 hr.	1 hr.	
4.21*	Remove Ferry Strap-on Engines	T_B + 2 hrs.	4.5 hrs.	
4.22	Position Emergency Equipment	T_B + 6.5 hrs.	15 min.	
4.23	Provide Access to Required Areas	T_B + 6.5 hrs.	30 min.	
4.24	Deservice ACS	T_B + 7 hrs.	7 hrs.	
4.25	Deservice Propulsion System	T_B + 14 hrs.	12.5 hrs.	
4.26	Deservice APU System	T_B + 14 hrs.	7 hrs.	
4.27	Deservice Fuel Cells	T_B + 14 hrs.	7 hrs.	
4.28	Move S/C to Pre-Flight Maintenance Area	T_B + 26.5 hrs.	1 hr.	
4.29	Install S/C on Handling/Transporting Vehicle	T_B + 27.5 hrs.	1 hr.	
4.30	Position Maintenance AGE	T_M	30 min.	$T_M = T_B + 28.5$ hrs.
4.31	Begin Quality Assurance Inspection	T_M + 30 min.	119.5 hrs.	Continuous
4.32	Checkout S/C Subsystems	T_M + 30 min.	4.5 hrs.	Maximum use is made of onboard checkout system (OCS).

* Applies to landing at remote site only.

Table 4-1 (9)

SEQUENCE OF EVENTS - LOGISTICS MISSION

	MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
4.33	Provide Access to Required Areas	T_M + 5 hrs.	8 hrs.	
4.34	Perform Scheduled Maintenance	T_M + 13 hrs.	80 hrs.	
4.35	Perform Unscheduled Maintenance	T_M + 13 hrs.	80 hrs.	Maximum time.
4.36	Close Access Areas	T_M + 93 hrs.	3 hrs.	
4.37	Perform Post-Maintenance Procedures	T_M + 96 hrs.	24 hrs.	
4.38	Load S/C on Erector/Transporter	T_M + 113 hrs.	1.5 hrs.	
4.39	Load Crew Provisions on S/C	T_M + 114.5 hrs.	30 min.	
4.40	Load Payload Canister on S/C	T_M + 115 hrs.	3 hrs.	Less passengers.
4.41	Load Jet Fuel	T_M + 118 hrs.	2 hrs.	

Begin PRELAUNCH OPERATIONS PHASE (See Phase 0.0)

Note that the on-pad prelaunch operations time is 24 hours. All intermediate events times were determined under the assumption of around-the-clock (3-shifts) ground crew and mission control operations. The prelaunch operations timeline is shown in Figure 4-2 with times corresponding to Table 4-1.

A further breakdown of the cryogenic servicing operation is shown in the timeline of Figure 4-3. Here, the times are computed on the basis of fuel and oxidizer loading beginning two hours before liftoff. In order to meet this groundrule, it is necessary to have parallel loading of both the drop tanks and the core vehicle together with parallel loading of both fuel and oxidizer, LH_2 and LO_2 in this case.

The post-flight maintenance and pre-flight readiness times of the Maintenance Phase reflect a short on-the-ground turnaround time of six to seven days. Again, three-shift operations are used to hold the total turnaround time to a minimum. For low launch rates, however, the entire maintenance task can easily be converted to the more economical one-shift no-weekend operation.

The transfer of the payload to the space station from the core vehicle is then accomplished either autonomously from aboard the payload or through use of a third vehicle, called the Space Tug. The Space Tug's primary purpose would be to dock to the payload and push or pull it to the space station, where it docks the payload onto the station. Details of the hardware, design, and mechanisms involved in payload transfer are beyond the scope of the study.

Additional information on the ascent and descent trajectories of the mission are presented in Section 3.0 of this volume.

4.3 Payload - Core Vehicle Integration - With the payload canister defined as a 15-foot-diameter, 60-foot-long cylinder, the advantages of minimizing and/or standardizing the payload core vehicle interface become at once apparent. By standardizing this interface, the payload becomes interchangeable with payloads of other missions. Mission-peculiar equipment can then be charged against and located within the payload canister itself. The payload is then, to a large degree, autonomous; that is, the payload has the capability of completing its mission independently of the core vehicle. The core vehicle supports the payload by ferrying it to and from orbit, providing attitude stabilization where necessary, and manual support, if required.

PRELAUNCH OPERATIONS TIMELINE

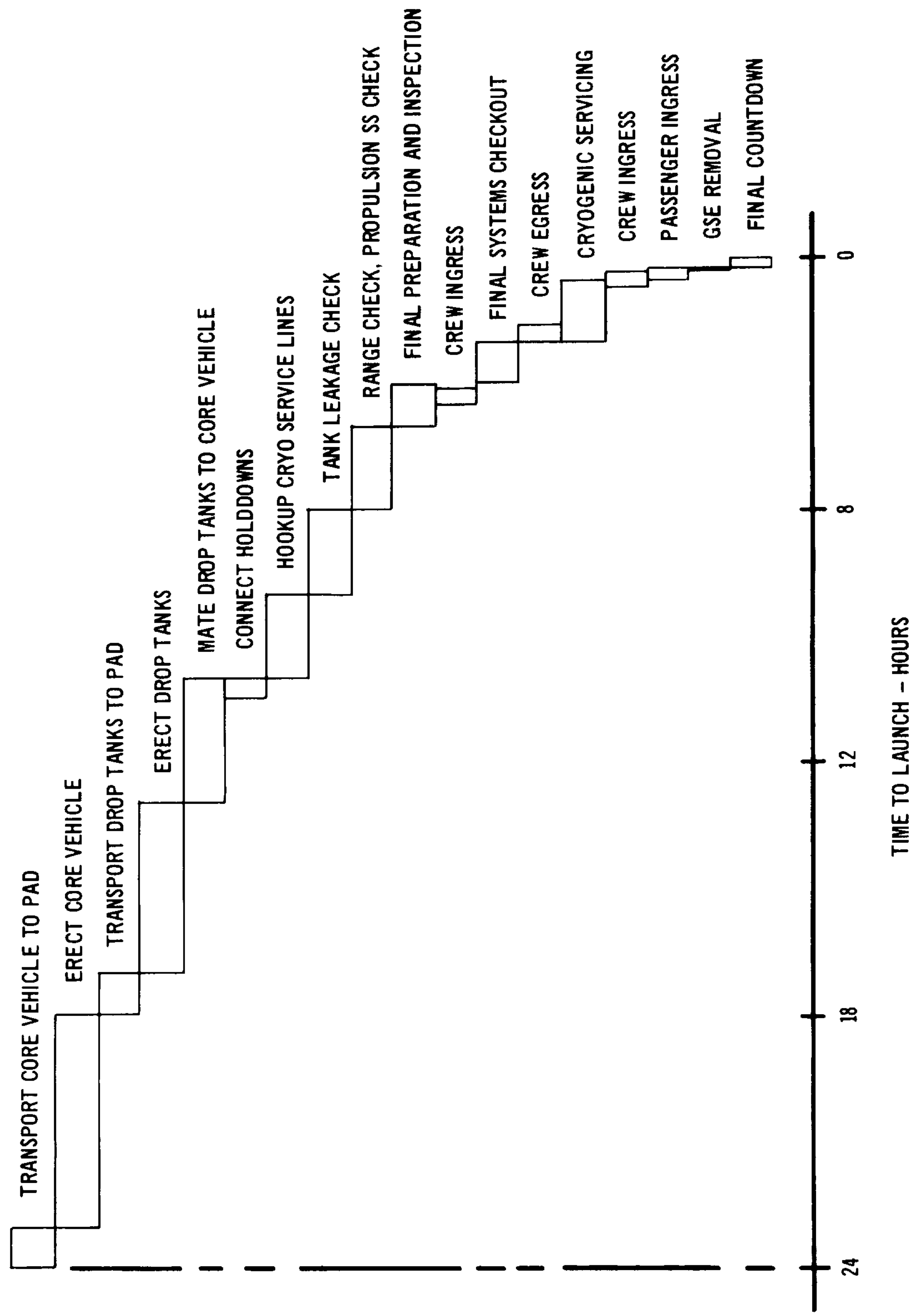


Figure 4-2

CRYOGENIC SERVICING TIMELINE

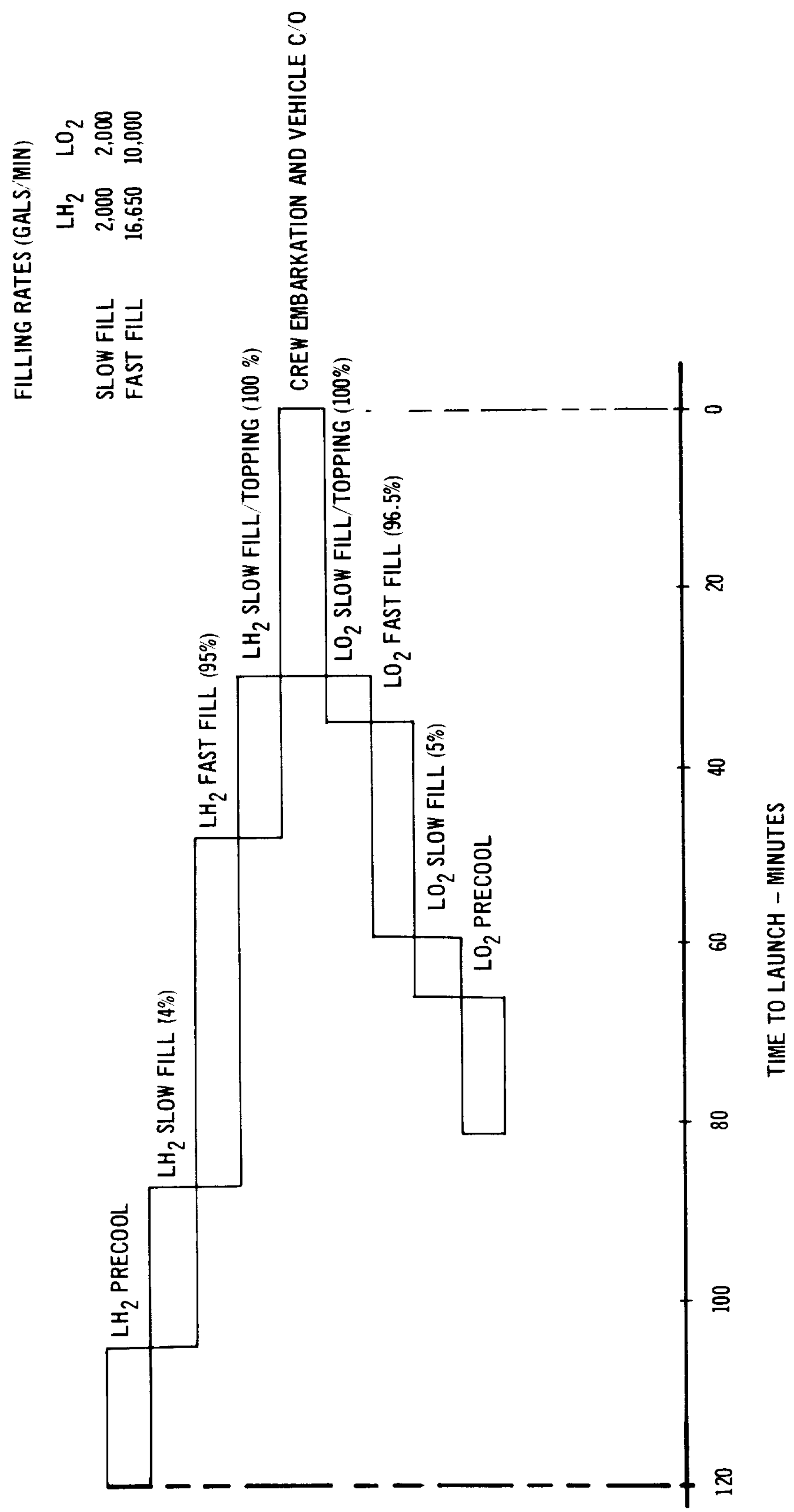


Figure 4-3

The potential ease with which the payloads of the integral payload of the ILRV concepts can be interchanged is illustrated in Figure 4-4. Although conceptually presented in this figure, alternate-mission payloads will be interchangeable with one another with only a minimal number of changes to the core vehicle, and with only short change-over times required. This is possible because the payload is first integrated within the mission module canister, the interface of the canister with the core vehicle is then maintained as simple as possible.

4.4 Payload Integration Modes - The use of an integral payload canister with the 1-1/2-stage ILRV concept requires a closer investigation of the spacecraft-payload interface and the integration of the payload into the payload canister. The first of these two areas is addressed in Section 4.3 above; the second is discussed below.

As illustrated in Figure 4-5, three payload integration modes are possible:

Mode I - This is the case where the payload canister is the mission module structure/avionics interface with the core vehicle and contains payload deployment mechanisms. In this case, the payload canister remains with the core vehicle whereas the payload itself does not.

Mode II - Here, the payload canister is used as the mechanism for deploying mission sensors. In this case, both the payload canister and its contents remain integral with the core vehicle.

Mode III - In this mode, the payload canister is the mission module and is left in orbit to operate autonomously.

A typical mission of the Mode I type is that of the delivery of advanced propulsive stages. This mission is illustrated in Figure 4-6 Satellite inspection/ maintenance also falls into this mode. Mode II is typified by the surveillance mission shown in the same figure. Missions of the logistic-resupply type fall in the Mode III category. Military-type missions may also fall under this category.

PAYLOAD INTERCHANGEABILITY

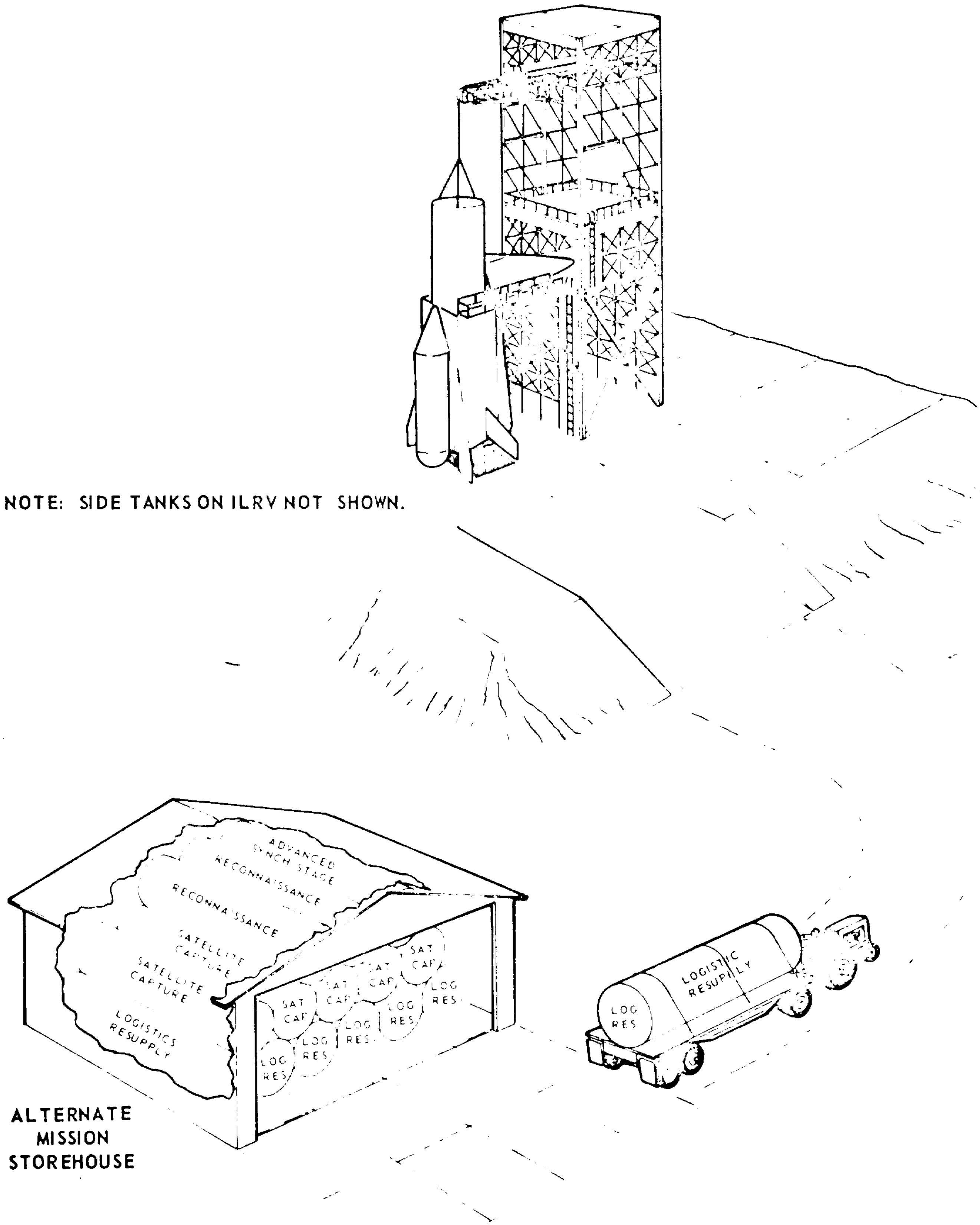


Figure 4–4

PAYLOAD INTEGRATION MODES

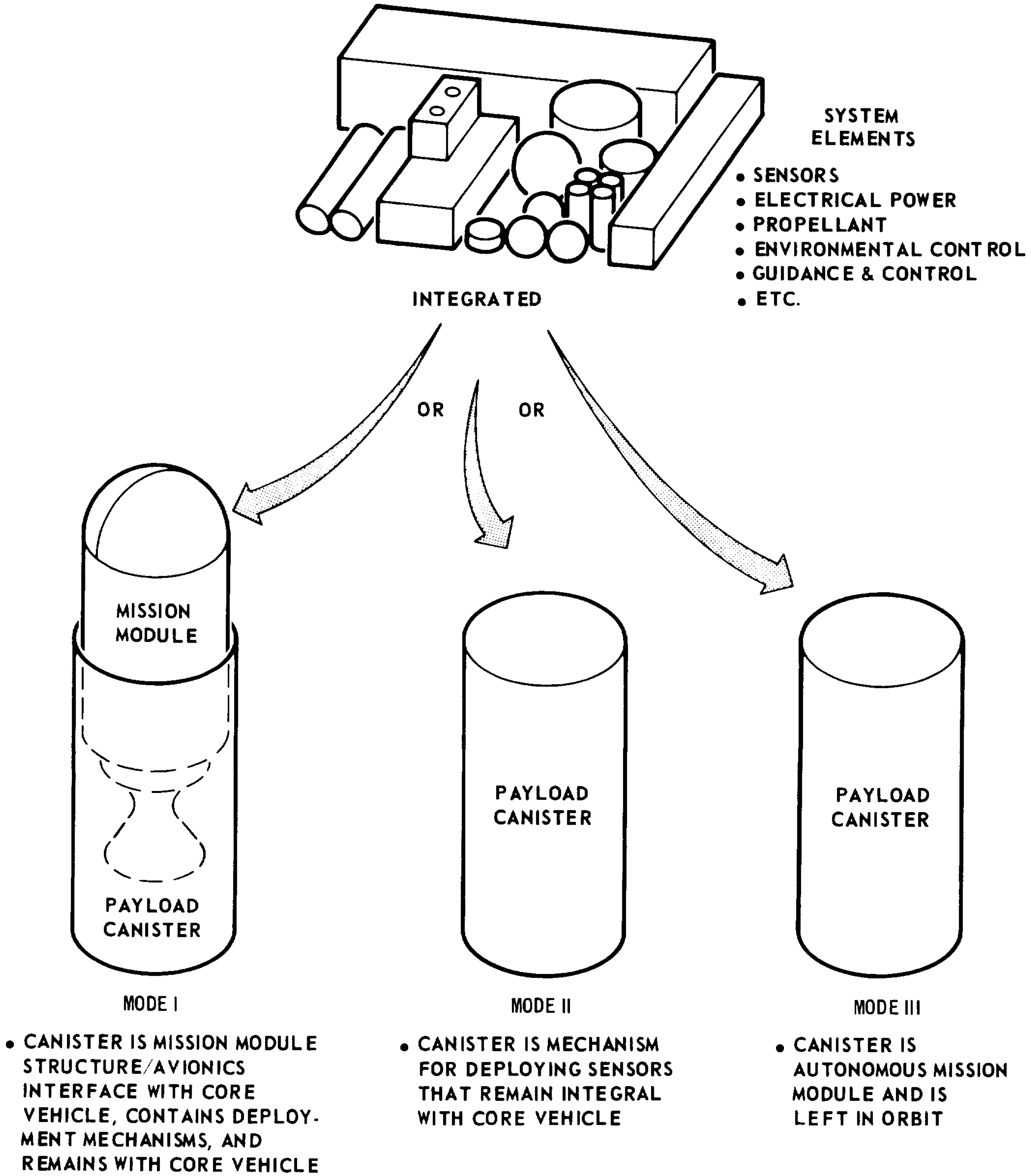


Figure 4-5

4.5 Swing-Nose Concept - One of the salient design features of the 1 1/2-stage ILRV configuration is the swing-nose concept, where the nose of the vehicle is swung away to allow the payload canister to be loaded and unloaded longitudinally. Vertical loading of the payload on the pad is illustrated in Figure 4-4. On-orbit payload loading/unloading is pictured in Figures 4-6 and 4-7.

On the pad, the swing-nose configuration allows the passengers to board and be seated in the horizontal position until nearly launch time. In orbit, the nose swings away to expose the payload which can then be docked with another vehicle or can be translated out from the core vehicle.

4.6 Alternate Missions - The ILRV mission profile for a logistic-resupply mission is shown in Figure 4-1. In addition to the resupply mission, alternate-mission capability is provided with the integral-payload-canister approach. Here, the alternate missions considered include:

- Delivery of advanced propulsive stages and/or payloads
- Satellite inspection and/or maintenance
- Space laboratory deployment/retrieval
- Earth resource surveillance

A short, simplified on-orbit mission profile for the accomplishment of each of the above-listed missions, together with the logistics mission, is illustrated in Figures 4-6 and 4-7.

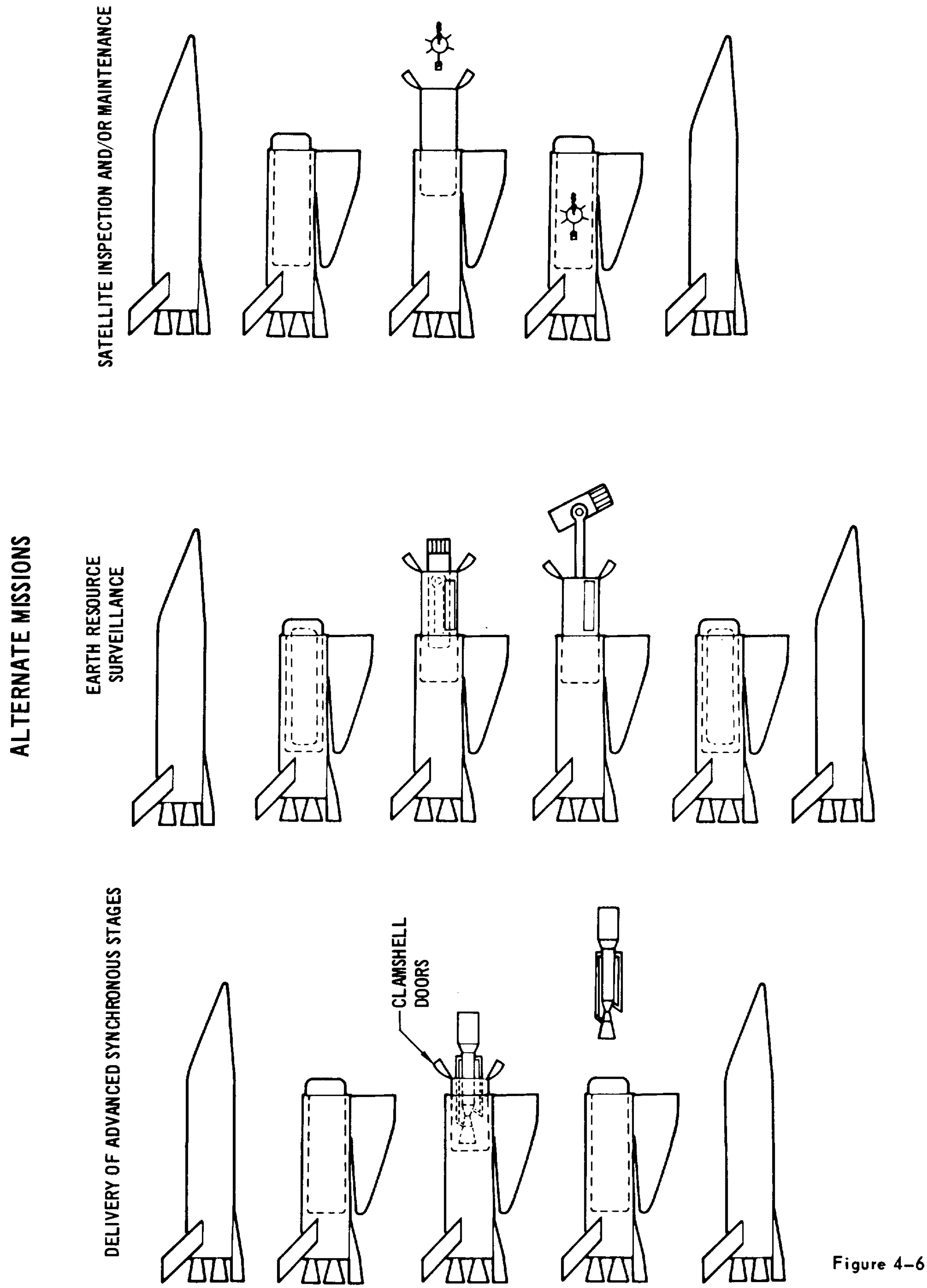

ALTERNATE MISSIONS
SATELLITE INSPECTION AND/OR MAINTENANCE
EARTH RESOURCE SURVEILLANCE
DELIVERY OF ADVANCED SYNCHRONOUS STAGES
CLAMSHELL DOORS

Figure 4-6

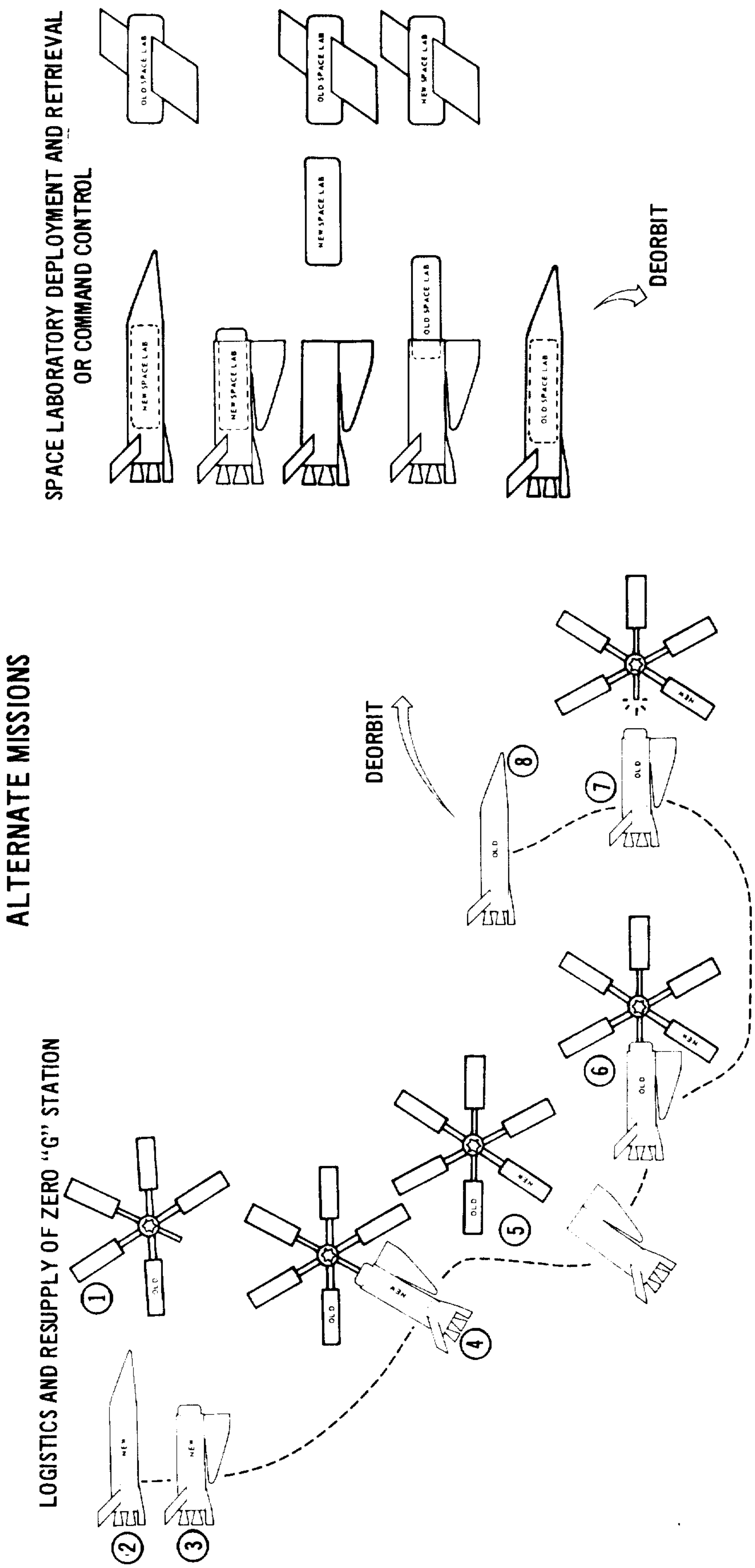

ALTERNATE MISSIONS
SPACE LABORATORY DEPLOYMENT AND RETRIEVAL OR COMMAND CONTROL
NEW SPACE LAB
OLD SPACE LAB
DEORBIT
LOGISTICS AND RESUPPLY OF ZERO "G" STATION
NEW
OLD
DEORBIT
1
2
3
4
5
6
7
8

Figure 4-7

5.0 VEHICLE AND PROGRAM COSTS

This section summarizes the results of the preliminary parametric cost analysis performed for the 1-1/2-stage design study which was terminated in early-August 1969.

The costs shown in this section are gross in that the vehicle, the development program and the operational programs were not defined to the depth and to the groundrules used in the later part of the study. It is therefore recommended that these costs not be used for comparison purposes between the 1-1/2-stage concept and the two-stage fully reusable concept.

The reader is directed to MDAC Report H367, "Integral Launch and Reentry System - Final Report", Volume I, 29 July 1969 for the preliminary cost analysis performed in conjunction with the following investigations.

5.1 Program Costs - A summary of the program cost estimates for six 1-1/2-stage ILRV configurations considered is given in Table 5-1. Configurations I, II, III, and IV were defined in Section 2.2. Configurations V and VI were extrapolated from that data for costing purposes.

The program costs are broken into RDT&E, investment, and operations costs and reflect the following groundrules:

- All costs are in millions of 1969 dollars.
- All costs are rough-order-of-magnitude (ROM) costs for planning purposes only.
- RDT&E costs are computed assuming two development vehicles and ten development flights.
- The program operational length is 10 years.
- The nominal (no-loss) launch rate is ten launches per year.
- The design life of the core vehicle is 30 uses.
- The launch-to-launch reliability of the core vehicle is .975.
- Production "learning" for the core vehicle is 95 percent; for the tip tanks, 90 percent.
- A spares factor of 10 percent is used for both core vehicle and tip tank investment.
- Fees are not included in the estimated costs.
- Launch operations costs are computed as a fixed percentage (15%) of the total hardware costs for that flight.

Table 5-1

MODEL 176M PROGRAM COST SUMMARY

- o Millions of 1969 Dollars
- o 10-Year Program Length
- o 10 Launches per Year

Configuration	I	II	III	IV	V	VI
Payload Weight (lbs.)	50,000	25,000	50,000	25,000	35,000	50,000
Payload Size (dia. x length)	15' x 60'	Unconstrained	22' x 26'	10' x 40'	12' x 48'	22' x 60'
Payload Volume (cu. ft.)						
Vehicle Length (ft.)	130	81.5	135	95.2	114.4	160
Gross Launch Weight (Millions of Pounds)	3.683	1.608	4.007	1.840	2.737	4.622
RDT&E Cost[1]	5,200	3,400	6,100	3,600	4,500	6,600
Investment Costs[2]	(1,975)	(1,237)	(2,231)	(1,331)	(1,673)	(2,367)
(1) Core Vehicle	1,193	810	1,391	861	1,045	1,485
(2) Tip Tanks	782	427	840	470	628	882
Operations Costs[3]	(1,853)	(1,243)	(2,147)	(1,327)	(1,609)	(2,352)
(1) Launch Operations	1,715	1,149	1,985	1,227	1,488	2,180
(2) Refurbishment	138	94	162	100	121	172
Total Program Cost	9,028	5,880	10,478	6,258	7,782	11,319

1 Includes Tip Tank Development Cost
2 Inventory for 105 Launches Plus Initial Spares
3 Launch Operations Cost = (.15) (1st unit cost of CV+1st unit cost of TT's) (105 Launches) (Learning Effect)
Refurbishment Cost = (.01) (1st unit cost of CV) (94 refurbishments) (Learning Effect)

- o Refurbishment costs are computed as a fixed percentage (1%) of the core vehicle hardware cost for that refurbishment.

5.2 Parametric Costs - The estimates of recurring cost for the six Model 176M configurations shown in Table 5-1 were investigated as functions of both launch operations and refurbishment fractions and are shown in Figures 5-1 through 5-6.

The recurring cost of a vehicle is the sum of its investment and operations costs. The launch operations fraction is defined as that percent (expressed as a decimal) of unit hardware cost which is assumed to estimate the launch operations costs for that vehicle. In this study, the launch operations fraction was varied from 0 to .15. The refurbishment fraction is defined as that percent (expressed as a decimal) of core vehicle unit hardware cost which is assumed to estimate the refurbishment cost for that vehicle. The refurbishment fraction is allowed to vary in steps of 0, .01, .03, .05, .08, .10, and .13. Note that the 0/0 fractions cost represents the total hardware investment cost for all the configurations.

- o Configuration I - The first unit costs are $206 million for the core vehicle and $5.6 million and $2.6 million for the larger and shorter pairs of tanks, respectively. The shaded area indicates a total recurring cost ranging from $3.80 to $5.0 billion. The corresponding average costs per pound of discretionary payload, developed over the 100-flight program, range from $760 to $1000.

- o Configuration II - The first unit cost for the core vehicle is $140 million and the first unit costs for the tip tank are $3.1 million for the larger pair and $1.3 million for the shorter pair. A total recurring cost ranging from $2.5 billion to $3.35 billion is indicated. The corresponding average recurring costs per pound of discretionary cargo delivered over the 100-flight program ranges from 1000 to about 1340 dollars.

- o Configuration III - The first unit cost of the core vehicle is $240 million, and the first unit costs of the tip tanks are $6.2 million and $2.5 million, respectively, for the longer and shorter pairs.

  The average cost per pound of discretionary payload delivered over the 100-flight program is seen to range from $870 to $1180.

TOTAL RECURRING COSTS

- MODEL 176M WITH 4 TIP TANKS
- LENGTH OF PROGRAM = 10 YEARS
- NUMBER OF SUCCESSFUL FLIGHTS = 100
- DESIGN LIFE OF E/V = 30 USES
- NUMBER OF E/V'S REQUIRED = 6
- E/V TURNAROUND TIME = 48–90 DAYS
- PROB. OF E/V RECOVERY = .975
- 15' X 60' CANISTER
- PAYLOAD = 50,000 LBS.

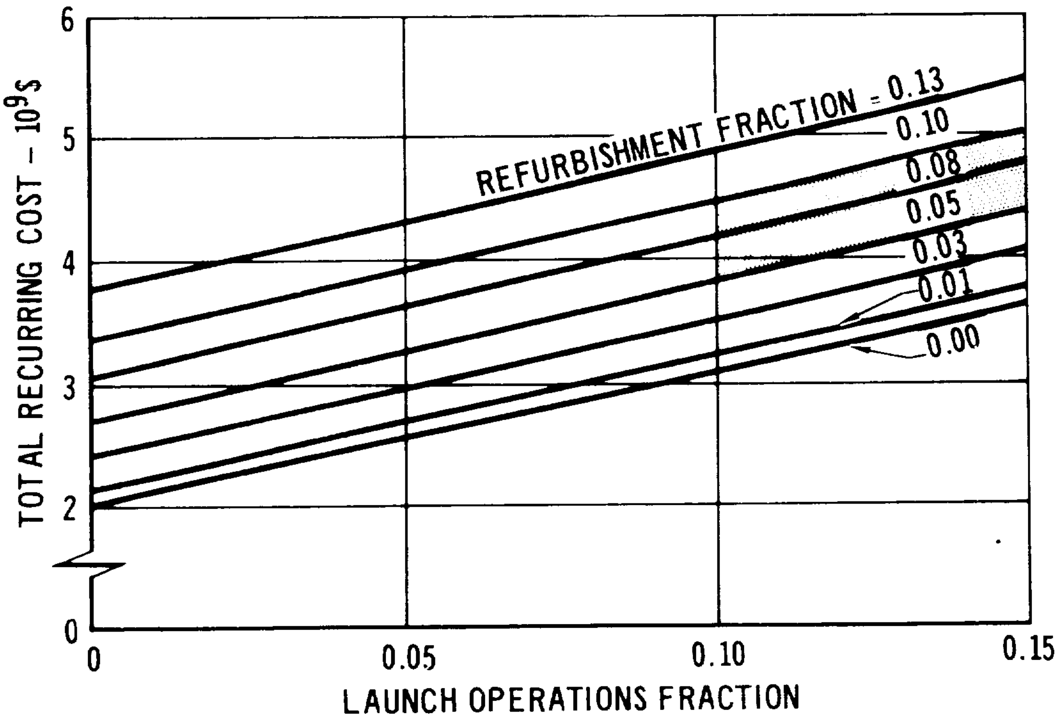


Figure 5-1

TOTAL RECURRING COSTS

- MODEL 176M WITH 4 TIP TANKS
- UNCONSTRAINED PAYLOAD VOLUME
- PAYLOAD = 25,000 LB
- LENGTH OF PROGRAM = 10 YEARS
- NUMBER OF SUCCESSFUL FLIGHTS = 100
- DESIGN LIFE OF E/V = 30 USES
- NUMBER OF E/V'S REQ'D = 6
- E/V TURNAROUND TIME = 48–90 DAYS
- PROBABILITY OF E/V RECOVERY = 0.975

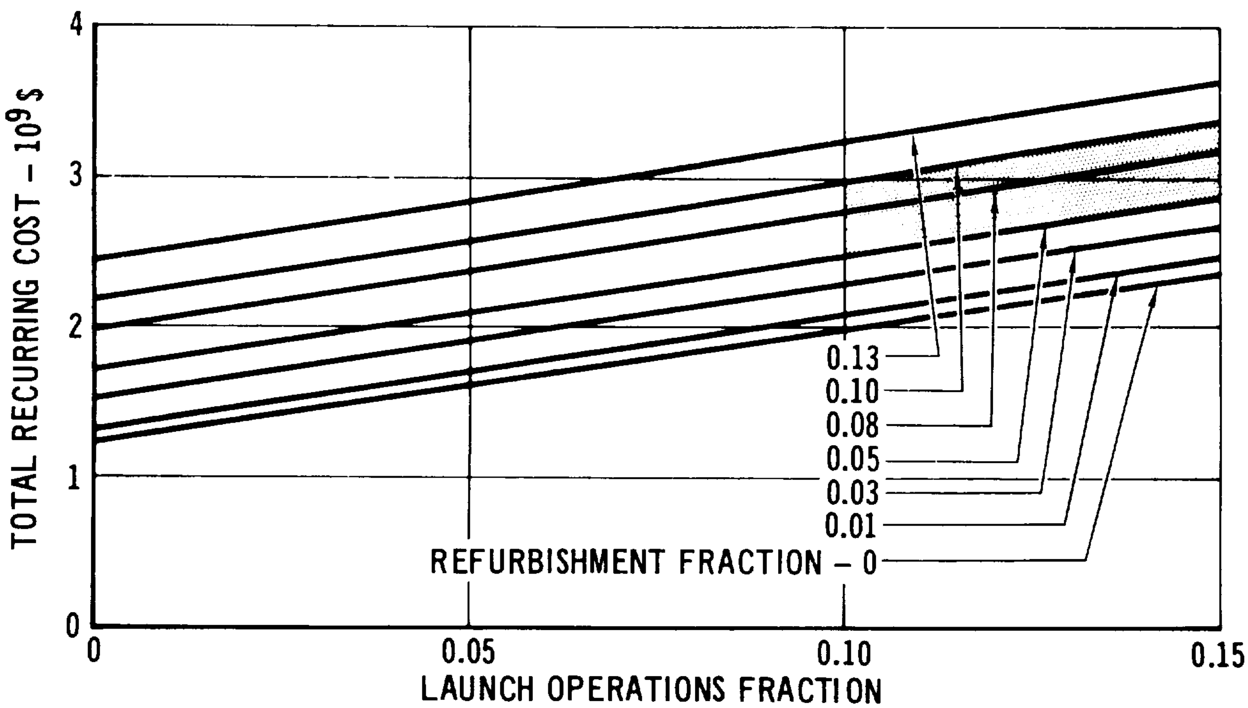


Figure 5–2

TOTAL RECURRING COSTS

- MODEL 176M WITH 4 TIP TANKS
- 22' x 26' PAYLOAD CANISTER
- PAYLOAD = 50,000 LB
- LENGTH OF PROGRAM 10 YEARS
- NUMBER OF SUCCESSFUL FLIGHTS = 100
- DESIGN LIFE OF E/V = 30 USES
- NUMBER OF E/V'S REQ'D = 6
- E/V TURNAROUND TIME = 48–90 DAYS
- PROBABILITY OF E/V RECOVERY = 0.975

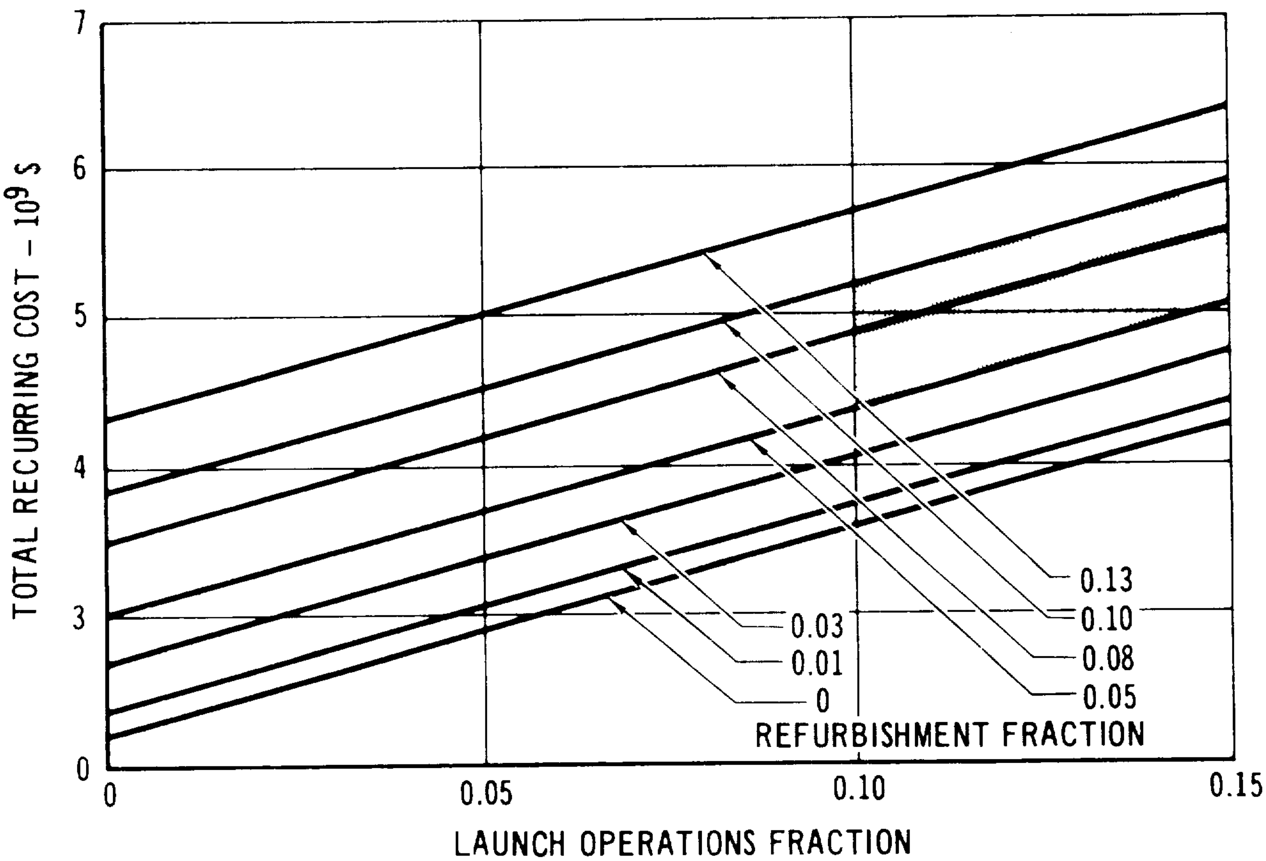


Figure 5–3

TOTAL RECURRING COSTS

- MODEL 176M WITH 4 TIP TANKS
- 10' x 40' PAYLOAD CANISTER
- PAYLOAD = 25,000 LB
- LENGTH OF PROGRAM = 10 YEARS
- NUMBER OF SUCCESSFUL FLIGHTS = 100
- DESIGN LIFE OF E/V = 30 USES
- NUMBER OF E/V'S REQ'D = 6
- E/V TURNAROUND TIME = 48–90 DAYS
- PROBABILITY OF E/V RECOVERY = 0.975

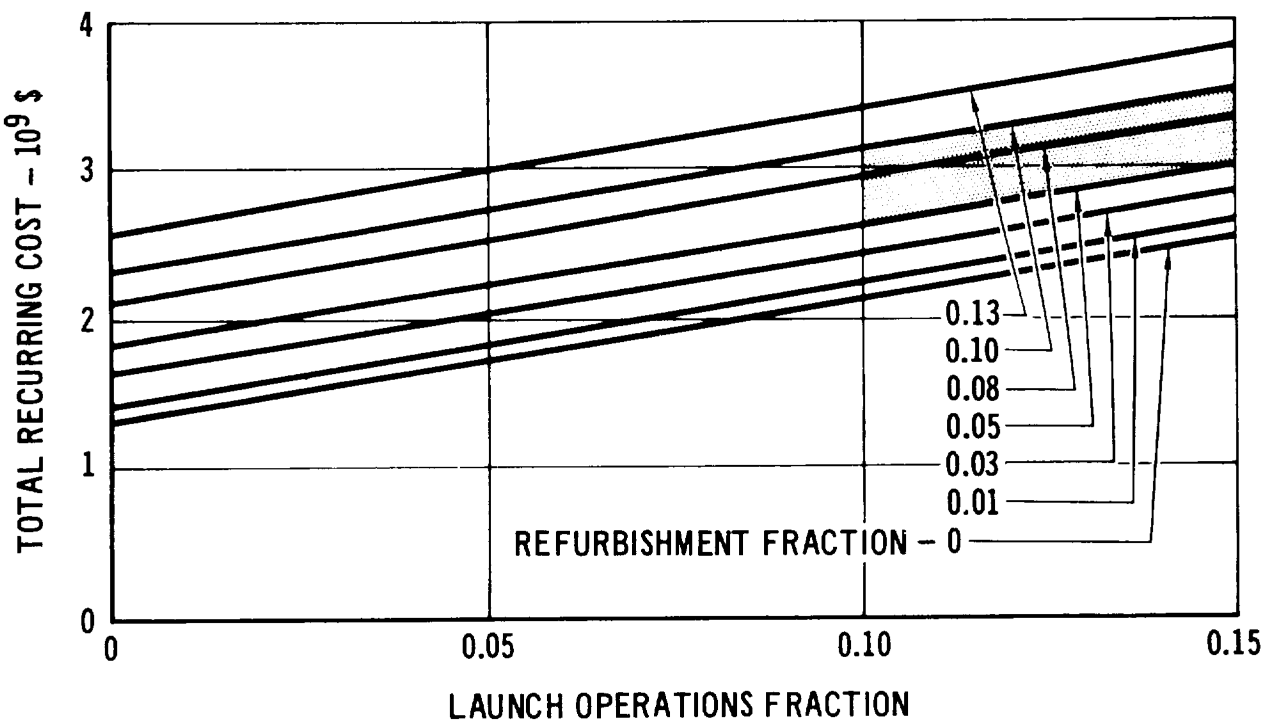


Figure 5-4

TOTAL RECURRING COSTS

- MODEL 176M WITH 4 TIP TANKS
- 12' x 48' PAYLOAD CANISTER
- PAYLOAD = 35,000 LB
- LENGTH OF PROGRAM = 10 YEARS
- NUMBER OF SUCCESSFUL FLIGHTS = 100
- DESIGN LIFE OF E/V = 30 USES
- NUMBER OF E/V'S REQ'D = 6
- E/V TURNAROUND TIME = 48–90 DAYS
- PROBABILITY OF E/V RECOVERY = 0.975

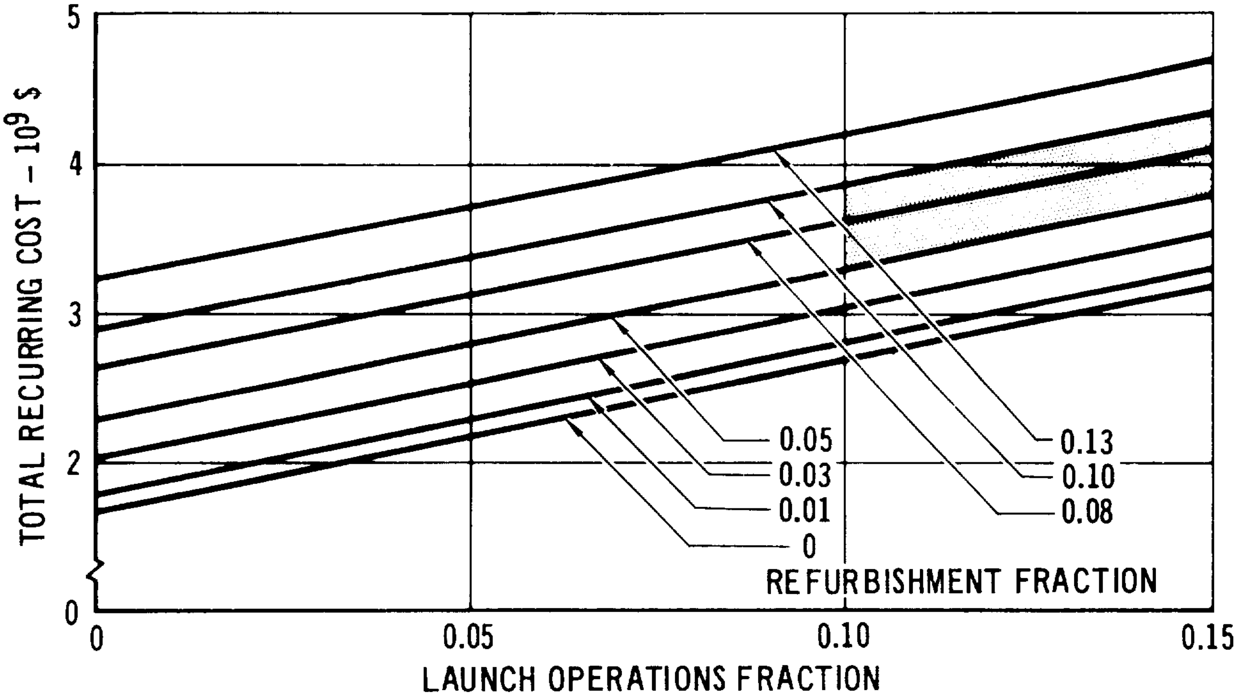


Figure 5–5

TOTAL RECURRING COST

- MODEL 176M WITH 4 TIP TANKS
- 22' x 60' PAYLOAD CANISTER
- PAYLOAD = 50,000 LB
- LENGTH OF PROGRAM = 10 YEARS
- NUMBER OF SUCCESSFUL FLIGHTS = 100
- DESIGN LIFE OF E/V = 30 USES
- NUMBER OF E/V'S REQ'D = 6
- E/V TURNAROUND TIME = 48–90 DAYS
- PROBABILITY OF E/V RECOVERY = 0.975

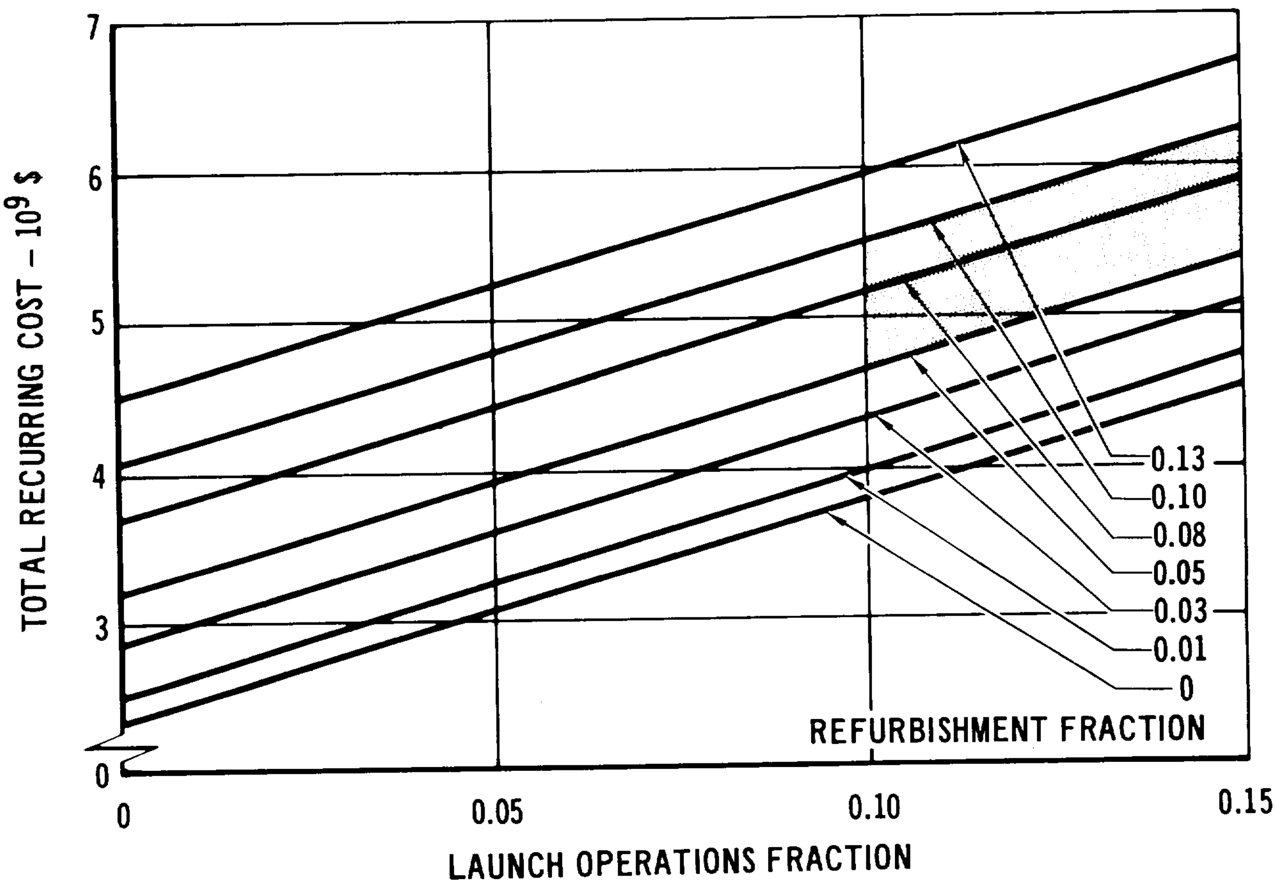


Figure 5-6

- Configuration IV - The first unit cost for the core vehicle is $149 million, the first unit costs for the tip tanks are $3.4 million for the longer pair and $1.45 million for the shorter pair. The area indicates a total recurring cost ranging from $2.65 billion to $3.55 billion.

  The corresponding average costs per pound of discretionary payload delivered over the 100-flight program range from $1060 to $1420. Nominally, the increase in average cost per pound of discretionary payload associated with the 10' x 40' payload dimensional constraint, as opposed to no dimensional constraints (Configuration II), is about $60 million to $80 million for the total 100-flight program.

- Configuration V - The total program cost over the region of interest ranges from about $3.3 to $4.35 billion. The corresponding costs per pound of discretionary payload, averaged over the 100-flight program, ranged from $945 to $1240.

- Configuration VI - The first unit cost of the core vehicle in this case is $256 million. For the tip tanks, the first unit costs run $6.4 million and $2.6 million for the longer and shorter tanks, respectively

  Over the 100-flight program, the average cost per pound of discretionary payload ranges from $930 to $1245.

5.3 Vehicle Cost Sensitivities - The variations of vehicle first unit and total program recurring costs with impulsive velocity distribution, propellant mass fraction, and propellant specific impulse for the four-tip-tank and the two-tip-tank 176M spacecraft, configured for the 15' x 60' canister containing 50,000 pounds of discretionary payload, are presented in Figure 5-7 through 5-21. The charts are arranged into three groups showing cost sensitivities to first-stage impulsive velocity, tank propellant mass fraction, and propellant specific impulse, respectively. The first chart in each group, e.g., Figures 5-7, 5-12, or 5-17, compares the total program recurring cost sensitivities of the two and four-tank configurations. Subsequent charts in each group provide the more detailed backup data used to derive the summary charts. In general, the cost of the four-tip-tank configuration is relatively insensitive to all three of the stated parameters. Further, at the nominal values of these parameters, the total program recurring costs for the four-tip-tank configuration is approximately $500 million less than that for the two-tip-tank configurations. The two-tip-tank configuration costs are shown to be very sensitive to the performance parameters. This is explained by the fact that the two-tip-tank configuration is more performance sensitive, i.e., tip-tank size (and thus, gross launch weight) are very sensitive to both tank propellant mass fraction and first-stage impulsive velocity.

TOTAL RECURRING COSTS SENSITIVITY TO ΔV DISTRIBUTION
Model 176M

- CARGO - 50,000 LB IN 15' x 60' CANISTER
- PROGRAM LENGTH = 10 YEARS
- NUMBER OF SUCCESSFUL FLIGHTS = 100
- DESIGN LIFE OF E/V = 30 USES
- NUMBER OF E/V'S REQ'D - 6
- PROB. OF E/V RECOVERY = 0.975
- S/C LEARNING - 95%
- TIP TANK LEARNING = 90%
- SPARES FACTOR = 10%
- LAUNCH OPNS - 10% OF 1ST UNIT HDWE COST
- REFURBISHMENT = 5% OF 1ST UNIT E/V COST

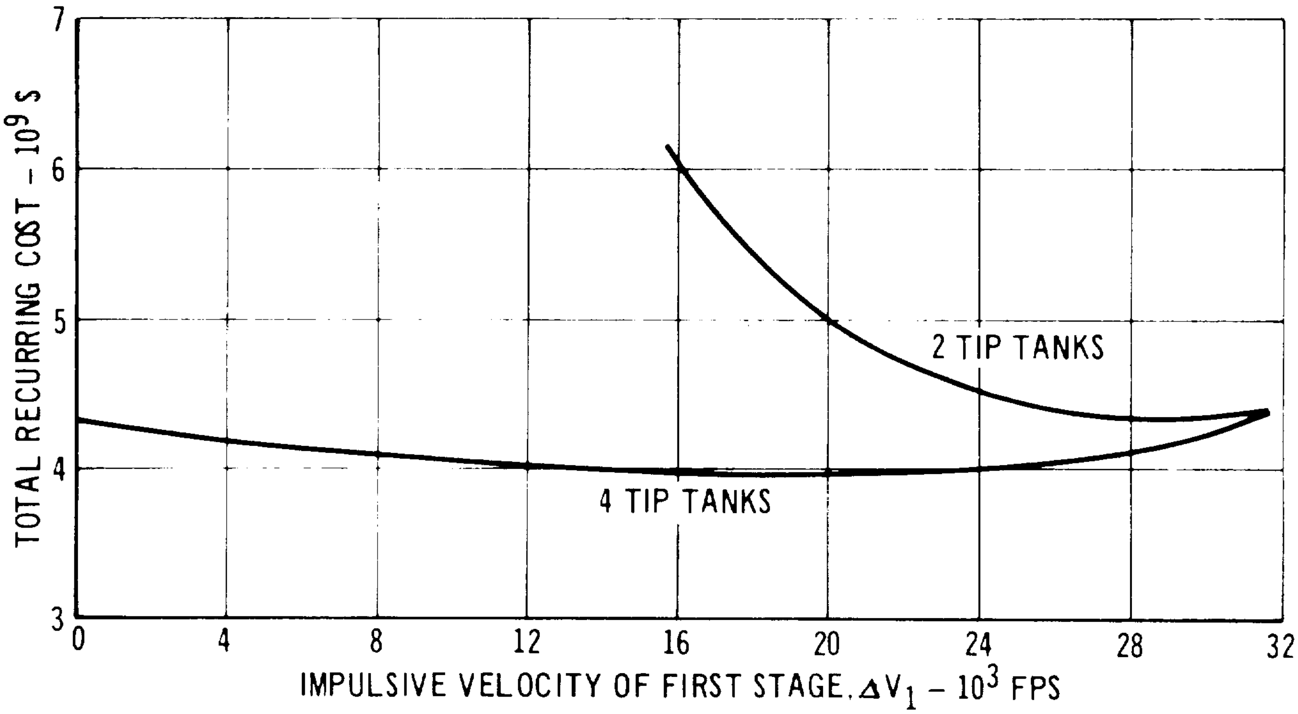


Figure 5-7

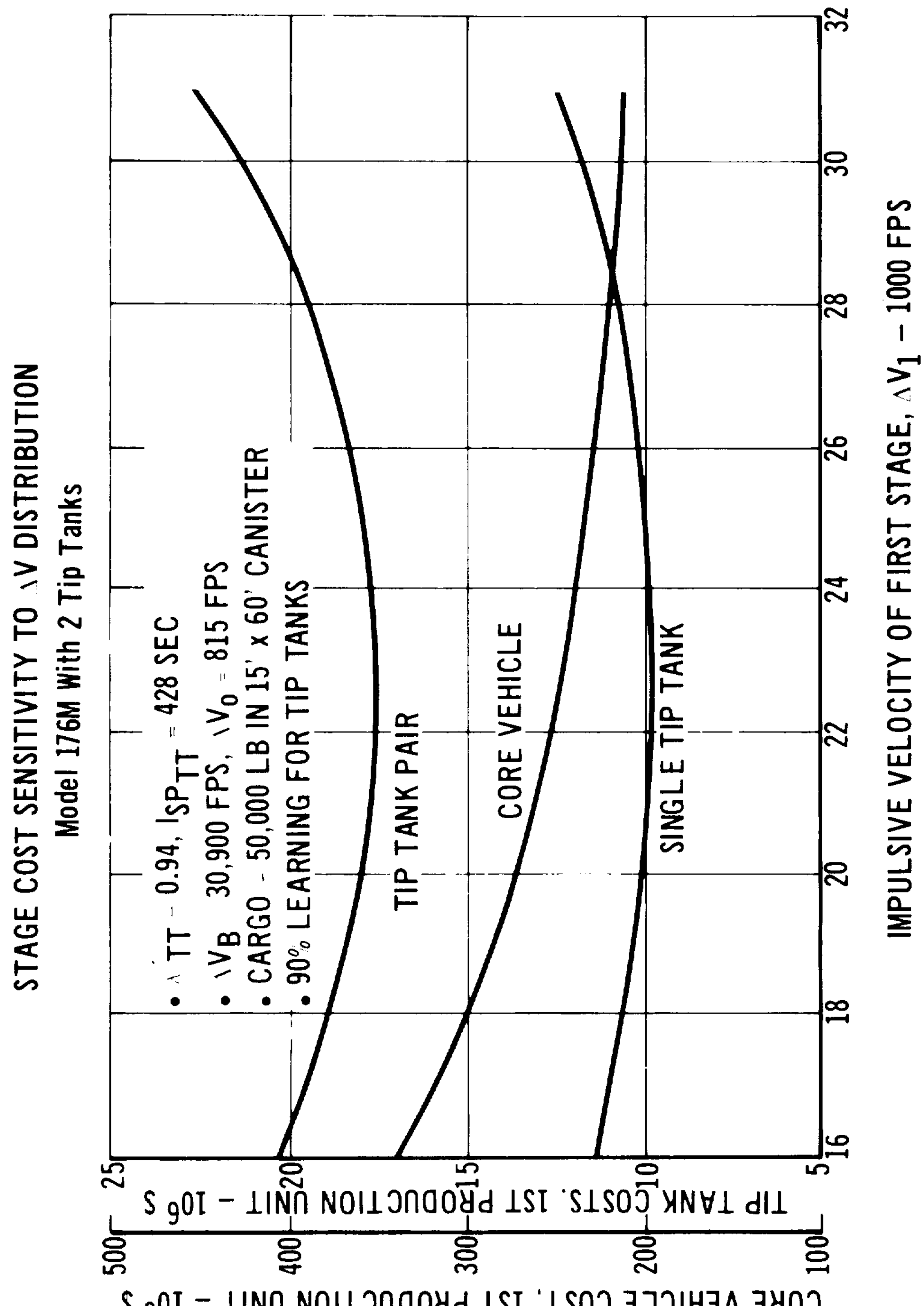

STAGE COST SENSITIVITY TO ΔV DISTRIBUTION
Model 176M With 2 Tip Tanks
• λ TT = 0.94, $I_{SP_{TT}}$ = 428 SEC
• ΔV_B 30,900 FPS, ΔV_0 = 815 FPS
• CARGO - 50,000 LB IN 15' x 60' CANISTER
• 90% LEARNING FOR TIP TANKS
TIP TANK PAIR
CORE VEHICLE
SINGLE TIP TANK
IMPULSIVE VELOCITY OF FIRST STAGE, ΔV_1 – 1000 FPS
16
18
20
22
24
26
28
30
32
TIP TANK COSTS, 1ST PRODUCTION UNIT – 10^6 \$
5
10
15
20
25
CORE VEHICLE COST, 1ST PRODUCTION UNIT – 10^6 \$
100
200
300
400
500

Figure 5-8

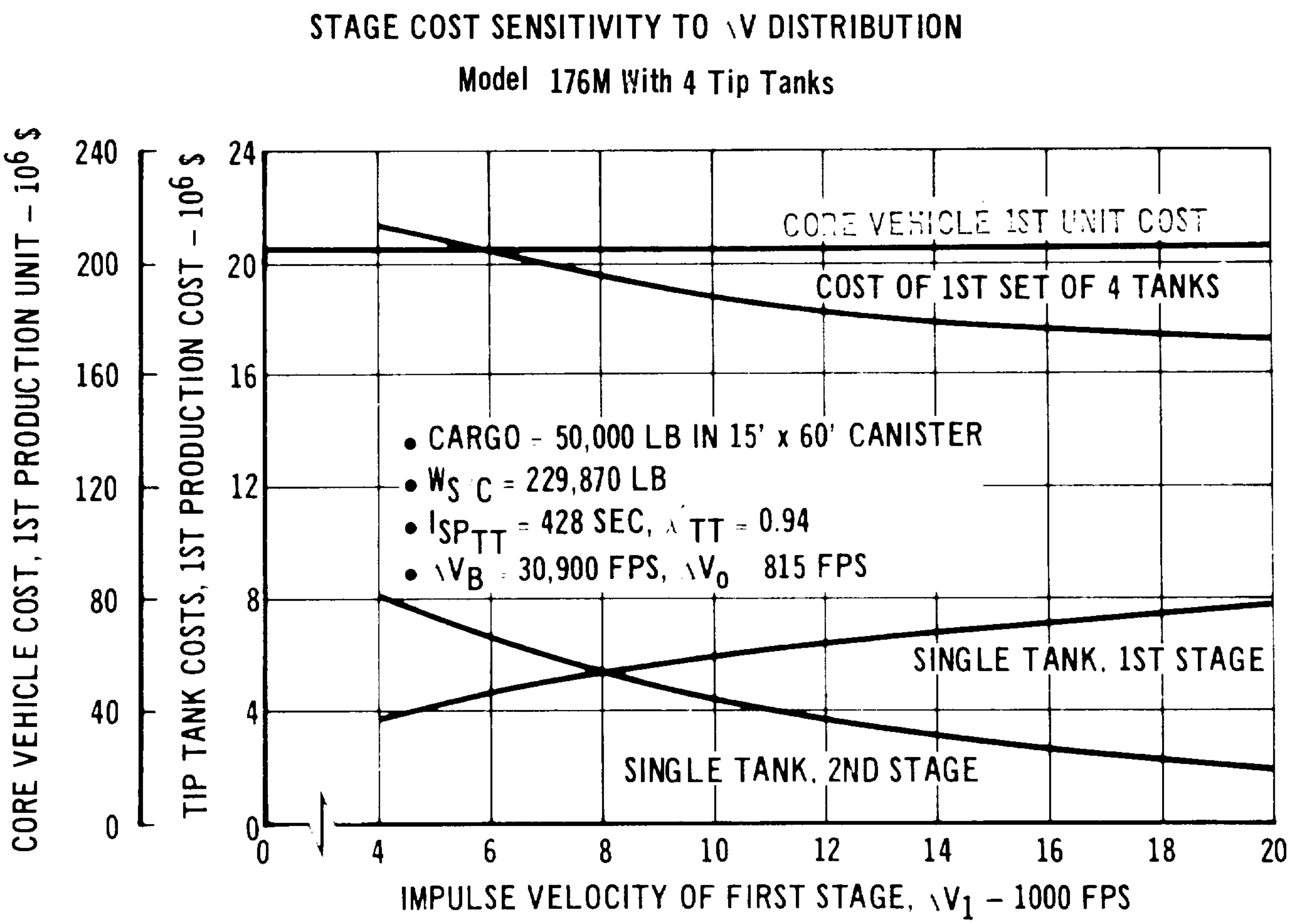

STAGE COST SENSITIVITY TO ∆V DISTRIBUTION
Model 176M With 4 Tip Tanks
CORE VEHICLE COST, 1ST PRODUCTION UNIT – 10^6 $
TIP TANK COSTS, 1ST PRODUCTION COST – 10^6 $
CORE VEHICLE 1ST UNIT COST
COST OF 1ST SET OF 4 TANKS
• CARGO = 50,000 LB IN 15' x 60' CANISTER
• $W_{S/C}$ = 229,870 LB
• $I_{SP_{TT}}$ = 428 SEC, λ'_{TT} = 0.94
• ΔV_B = 30,900 FPS, ΔV_0 815 FPS
SINGLE TANK, 1ST STAGE
SINGLE TANK, 2ND STAGE
IMPULSE VELOCITY OF FIRST STAGE, ΔV_1 – 1000 FPS
240
200
160
120
80
40
0
24
20
16
12
8
4
0
0
4
6
8
10
12
14
16
18
20

Figure 5-9

RECURRING COST SENSITIVITY TO ΔV DISTRIBUTION
Model 176M With 4 Tip Tanks

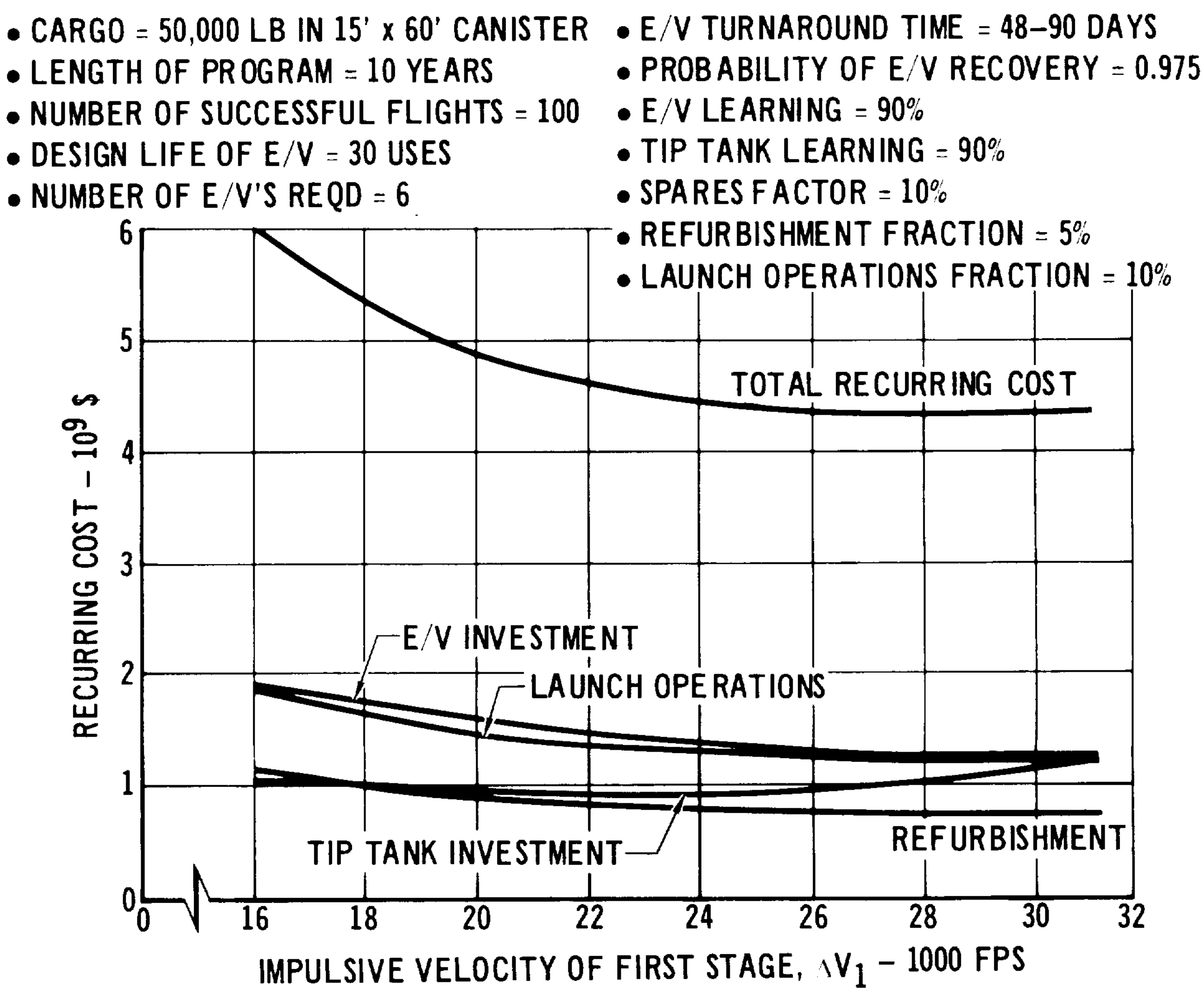


Figure 5–10

RECURRING COST SENSITIVITY TO ΔV DISTRIBUTION
Model 176M With 2 Tip Tanks

- CARGO = 50,000 LB IN 15' x 60' CANISTER
- $W_{S/C}$ = 229,870 LB
- PROGRAM LENGTH = 10 YEARS
- NUMBER OF SUCCESSFUL FLIGHTS = 100
- DESIGN LIFE = 30 USES
- NUMBER OF E/V'S REQ'D = 6
- E/V TURNAROUND TIME - 48–90 DAYS
- PROBABILITY OF E/V RECOVERY = 0.975
- S/C LEARNING = 95%
- TIP TANK LEARNING = 90%
- SPARES FACTOR = 10%
- LAUNCH OPNS = 10% 1ST UNIT HDWE COST
- REFURBISHMENT = 5% 1ST UNIT S/C COST

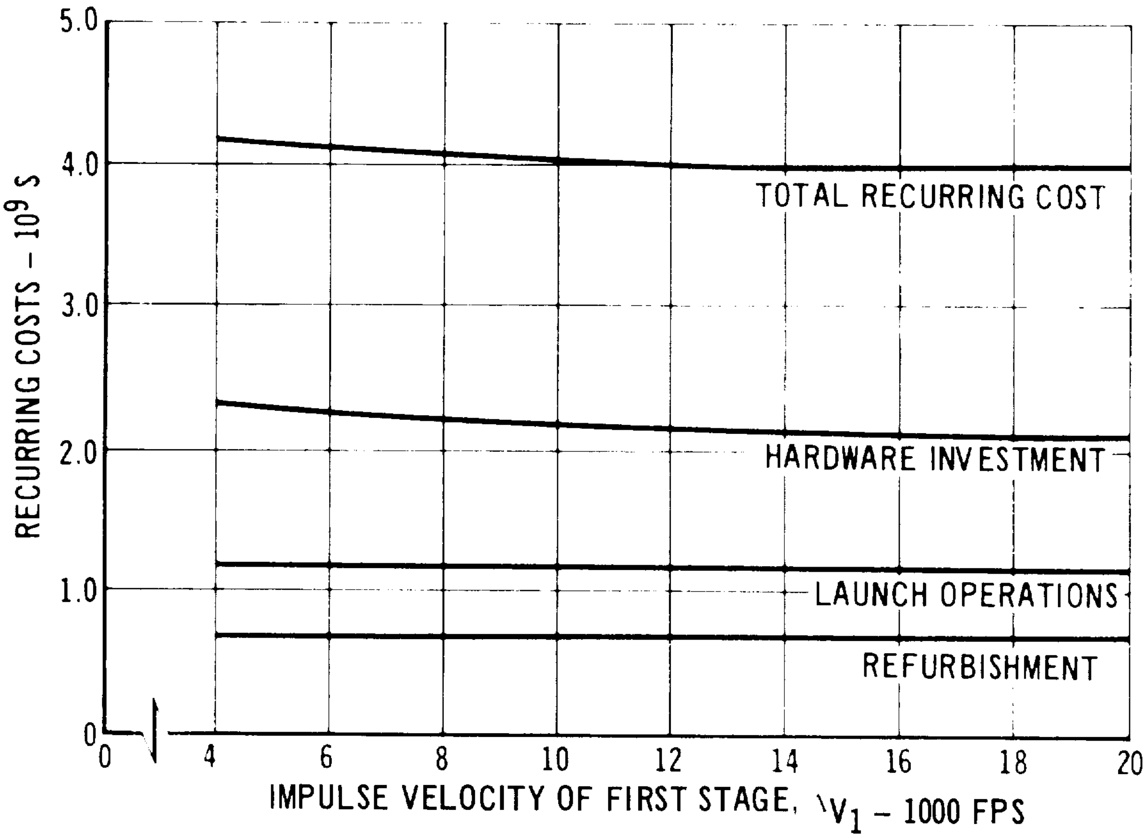


Figure 5–11

TOTAL RECURRING COST SENSITIVITY TO TANK PROPELLANT MASS FRACTION
Model 176M

- CARGO = 50,000 LB IN 15' x 60' CANISTER
- LENGTH OF PROGRAM = 10 YEARS
- NUMBER OF SUCCESSFUL FLIGHTS = 100
- DESIGN LIFE OF E/V = 30 USES
- NUMBER OF E/V'S REQ'D = 6
- E/V TURNAROUND TIME = 48–90 DAYS
- PROB. OF E/V RECOVERY = 0.975
- E/V LEARNING = 95%
- TIP TANK LEARNING = 90%
- SPARES FACTOR = 10%
- REFURBISHMENT FRACTION = 5%
- LAUNCH OPERATIONS FRACTION = 10%

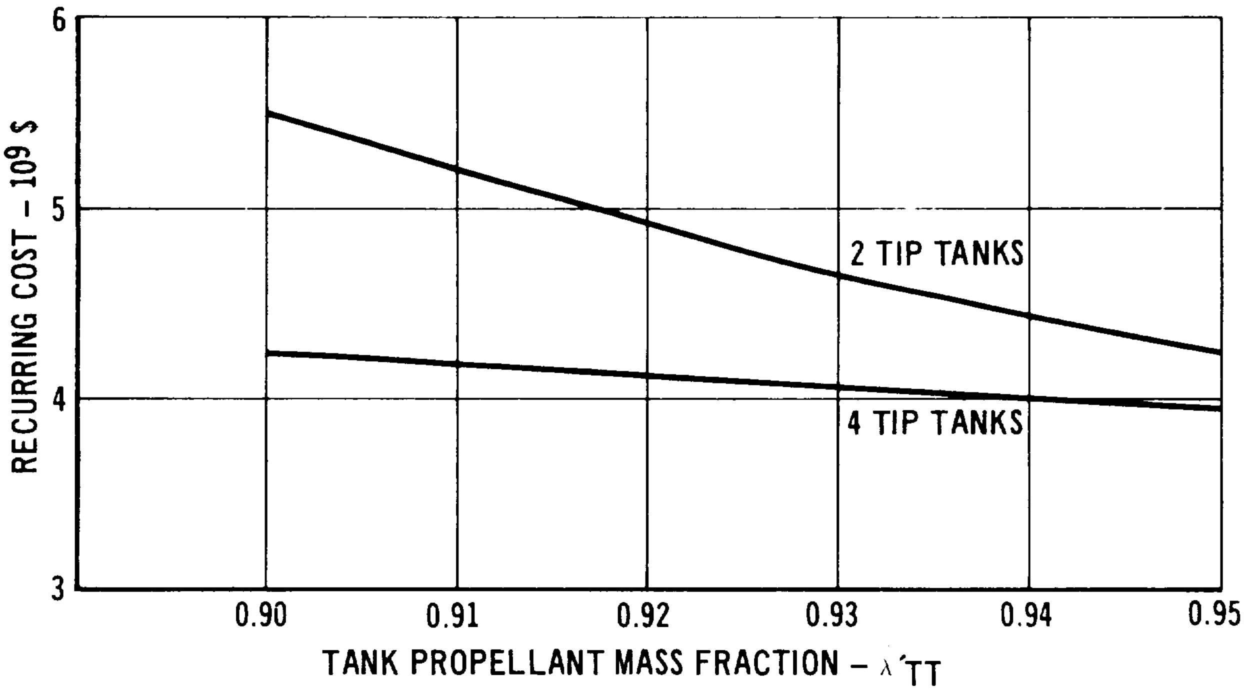


Figure 5–12

STAGE COST SENSITIVITY TO TANK PROPELLANT MASS FRACTION
Model 176M With 2 Tip Tanks

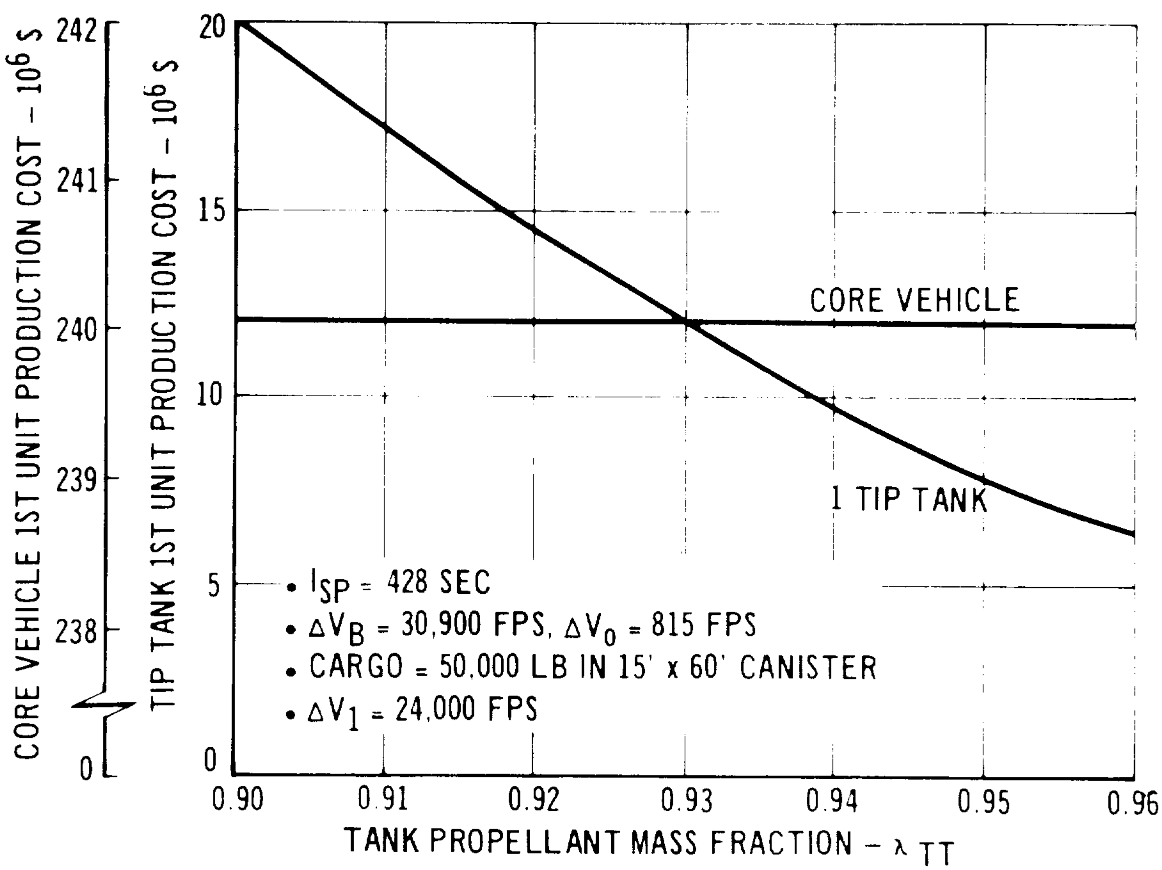


Figure 5-13

STAGE COST SENSITIVITY TO TANK PROPELLANT MASS FRACTION

Model 176M With 4 Tip Tanks

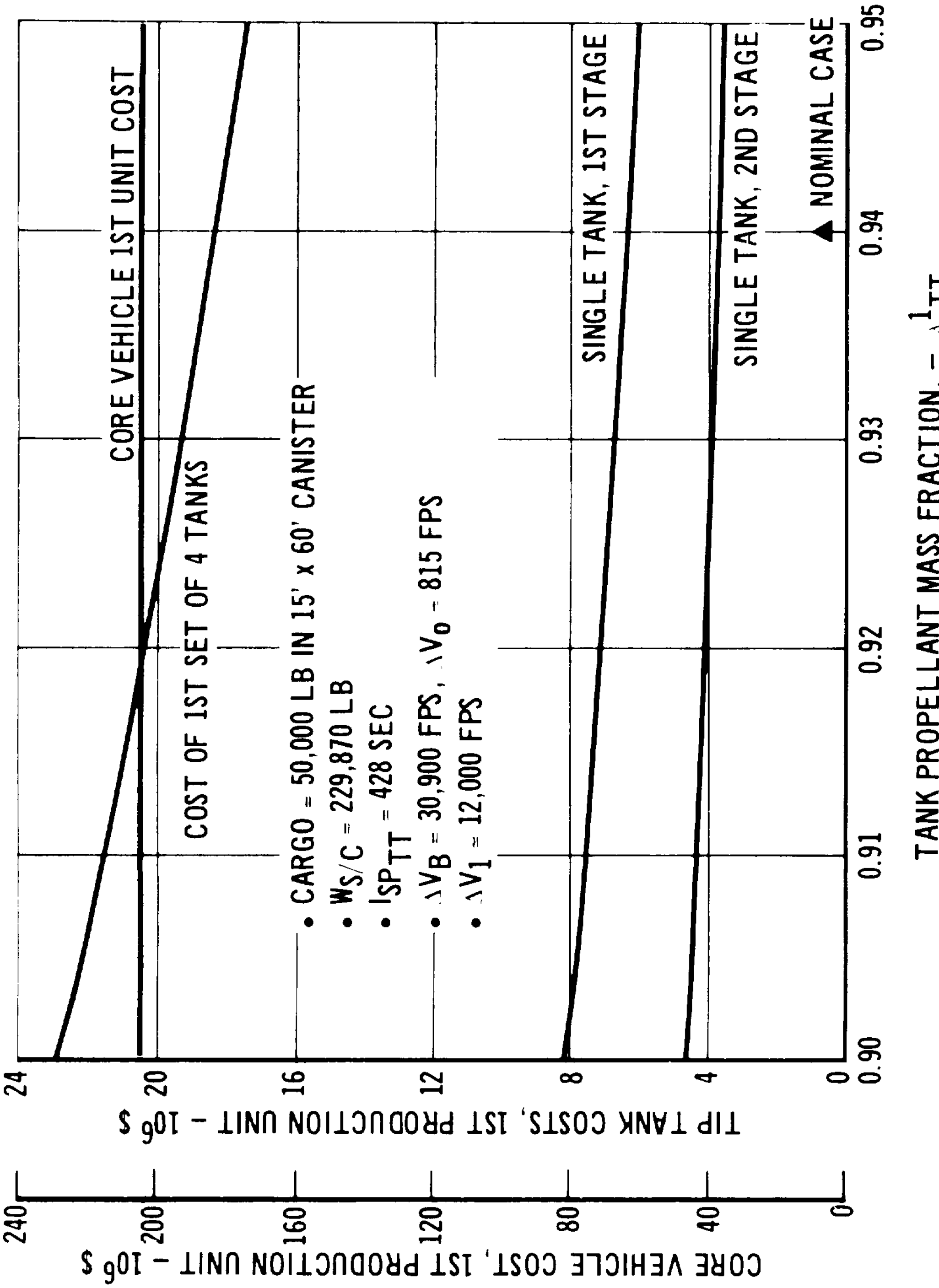


Figure 5–14

RECURRING COST SENSITIVITY TO TANK PROPELLANT MASS FRACTION
Model 176M With 4 Tip Tanks

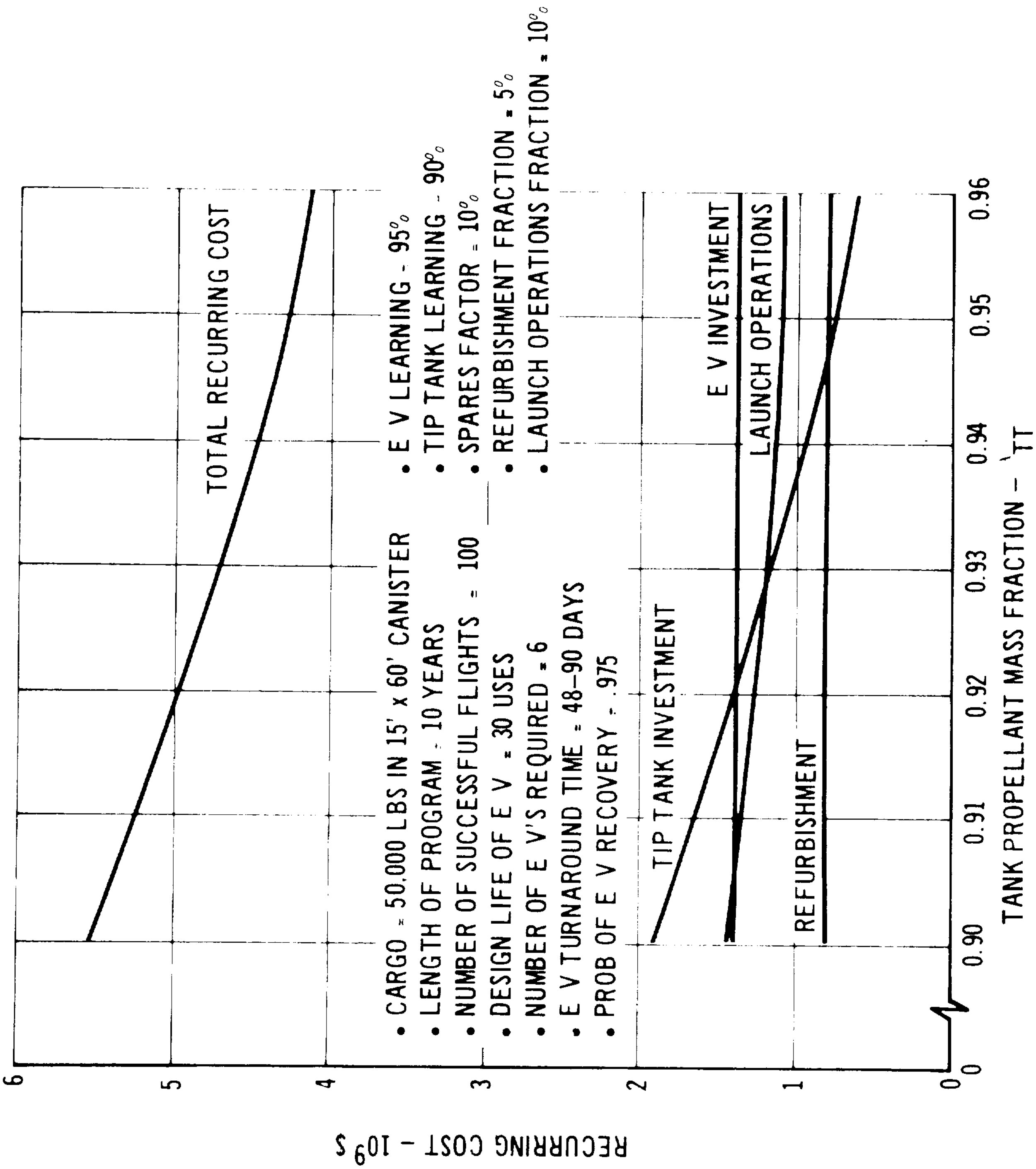


Figure 5-15

RECURRING COST SENSITIVITY TO TANK PROPELLANT MASS FRACTION

Model 176M With 4 Tip Tanks

- CARGO = 50,000 LBS IN 15' x 60' CANISTER
- $W_{S/C}$ = 229,870 LBS
- PROGRAM LENGTH = 10 YEARS
- NUMBER OF SUCCESSFUL FLIGHTS = 100
- DESIGN LIFE = 30 USES

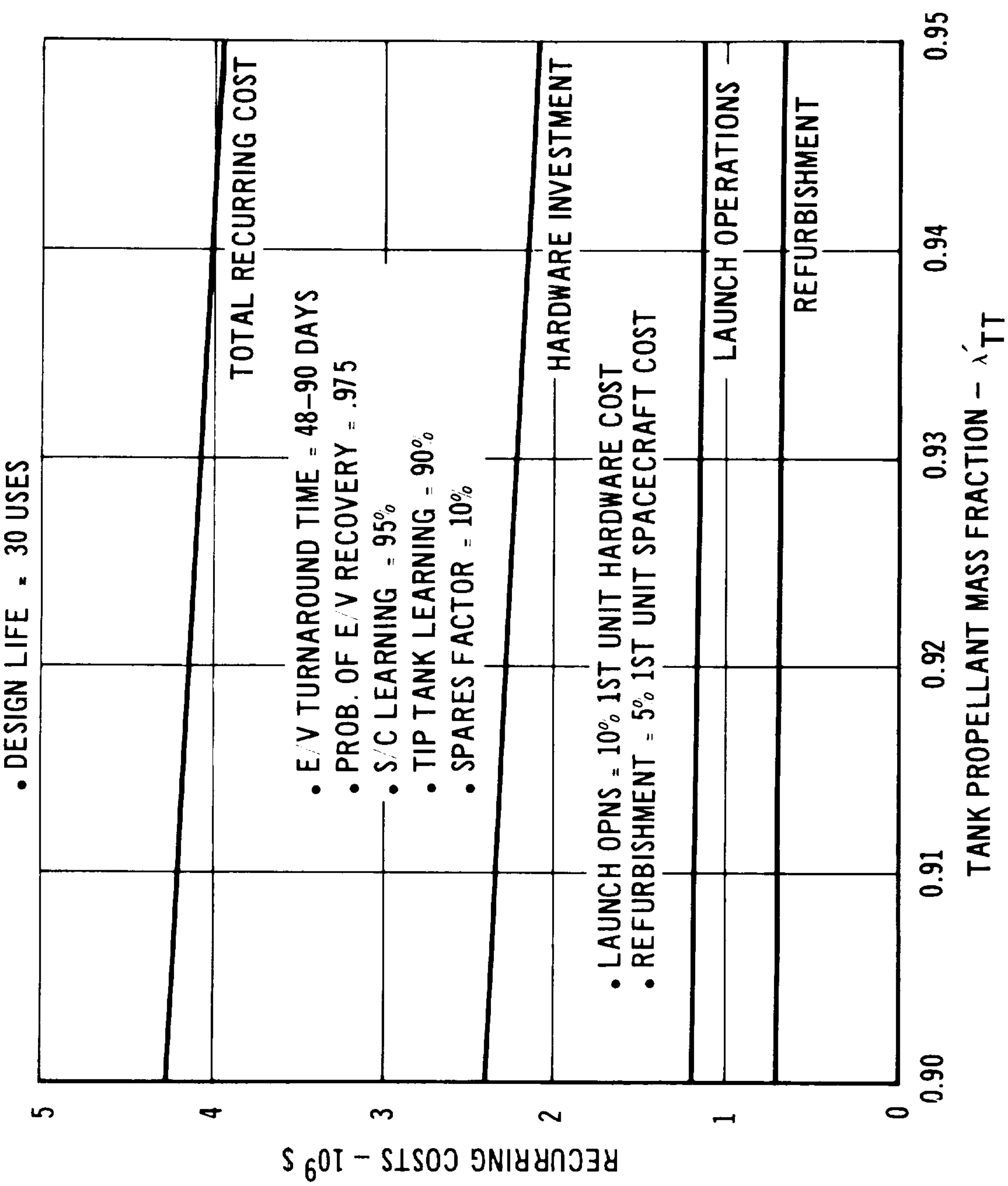


Figure 5–16

TOTAL RECURRING COST SENSITIVITY TO PROPELLANT SPECIFIC IMPULSE
Model 176M

- CARGO = 50,000 LB IN 15' x 60' CANISTER
- LENGTH OF PROGRAM = 10 YEARS
- NUMBER OF SUCCESSFUL FLIGHTS = 100
- DESIGN LIFE OF E/V = 30 USES
- NUMBER OF E/V'S REQ'D = 6
- E/V TURNAROUND TIME = 48–90 DAYS
- PROB. OF E/V RECOVERY = 0.975
- E/V LEARNING = 95%
- TIP TANK LEARNING = 90%
- SPARES FACTOR = 5%
- LAUNCH OPERATIONS FRACTION = 10%

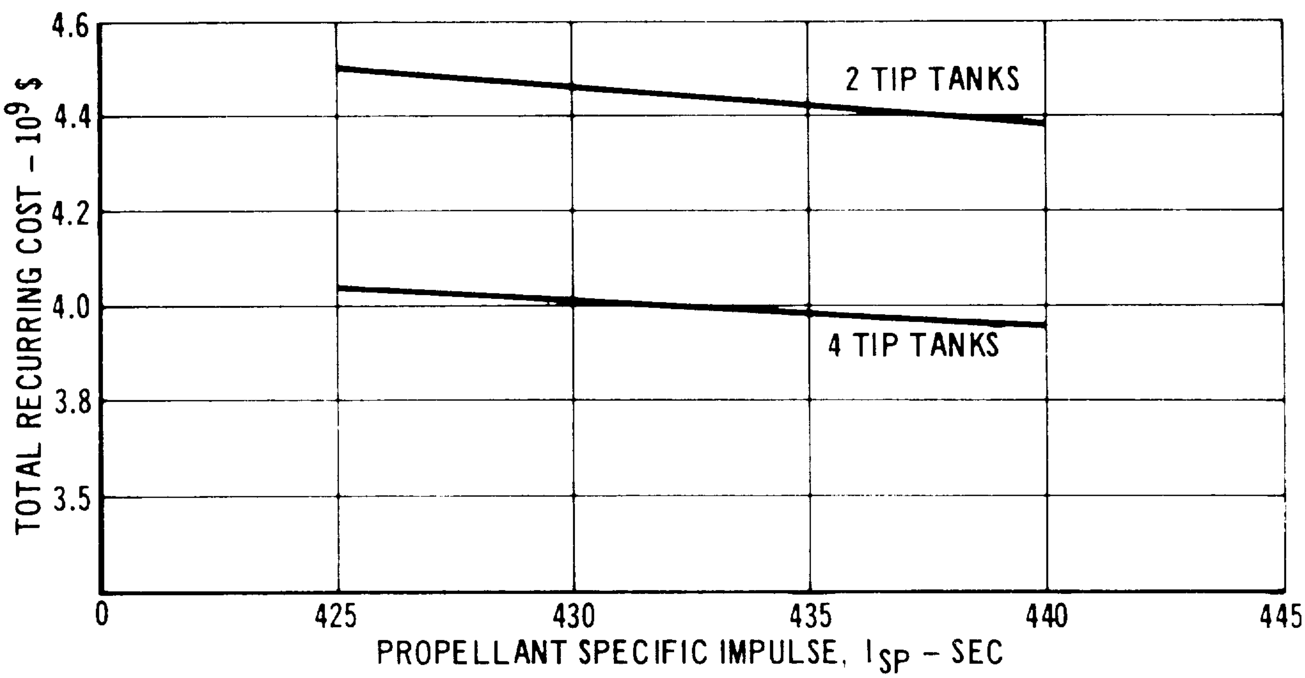
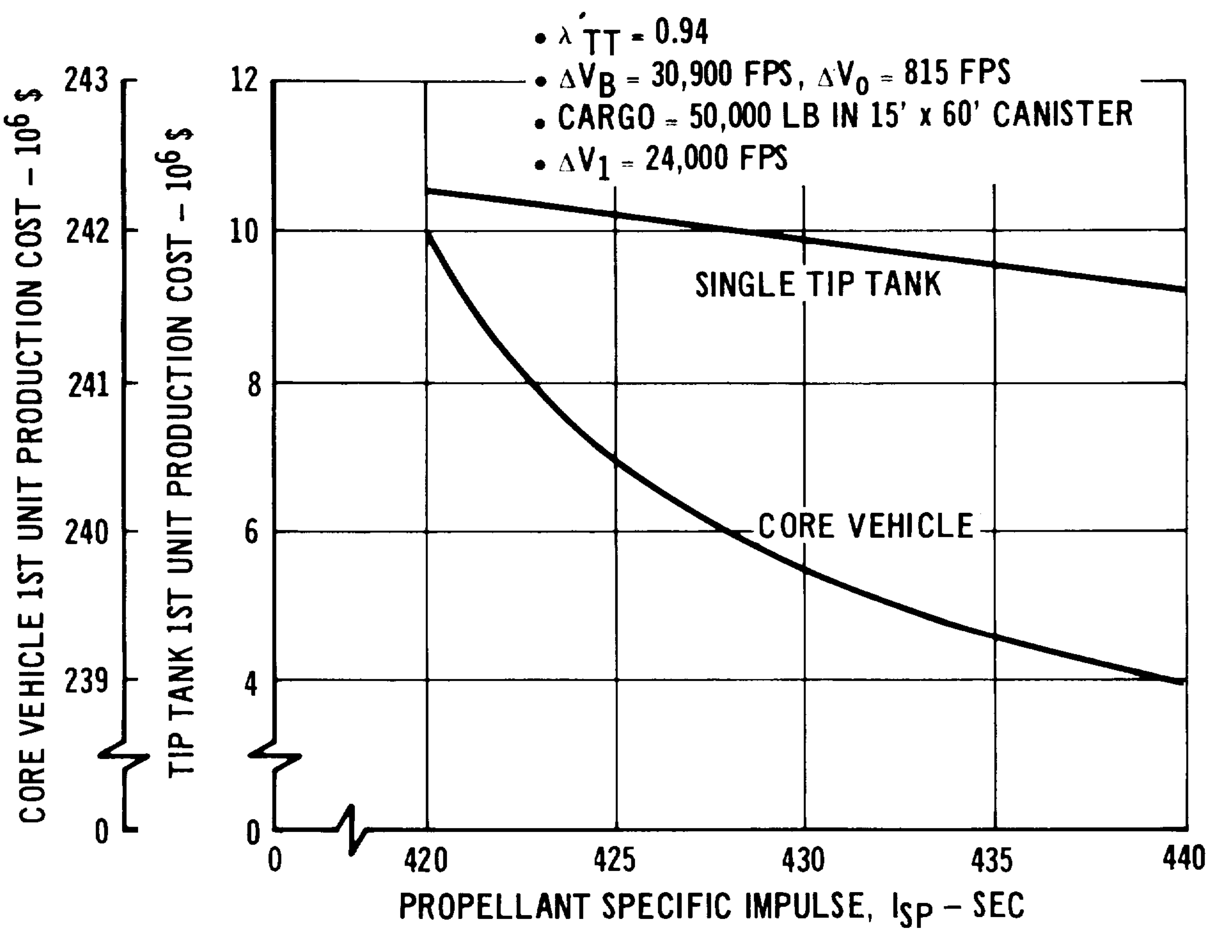


Figure 5–17

STAGE COST SENSITIVITY TO PROPELLANT SPECIFIC IMPULSE
Model 176M With 2 Tip Tanks



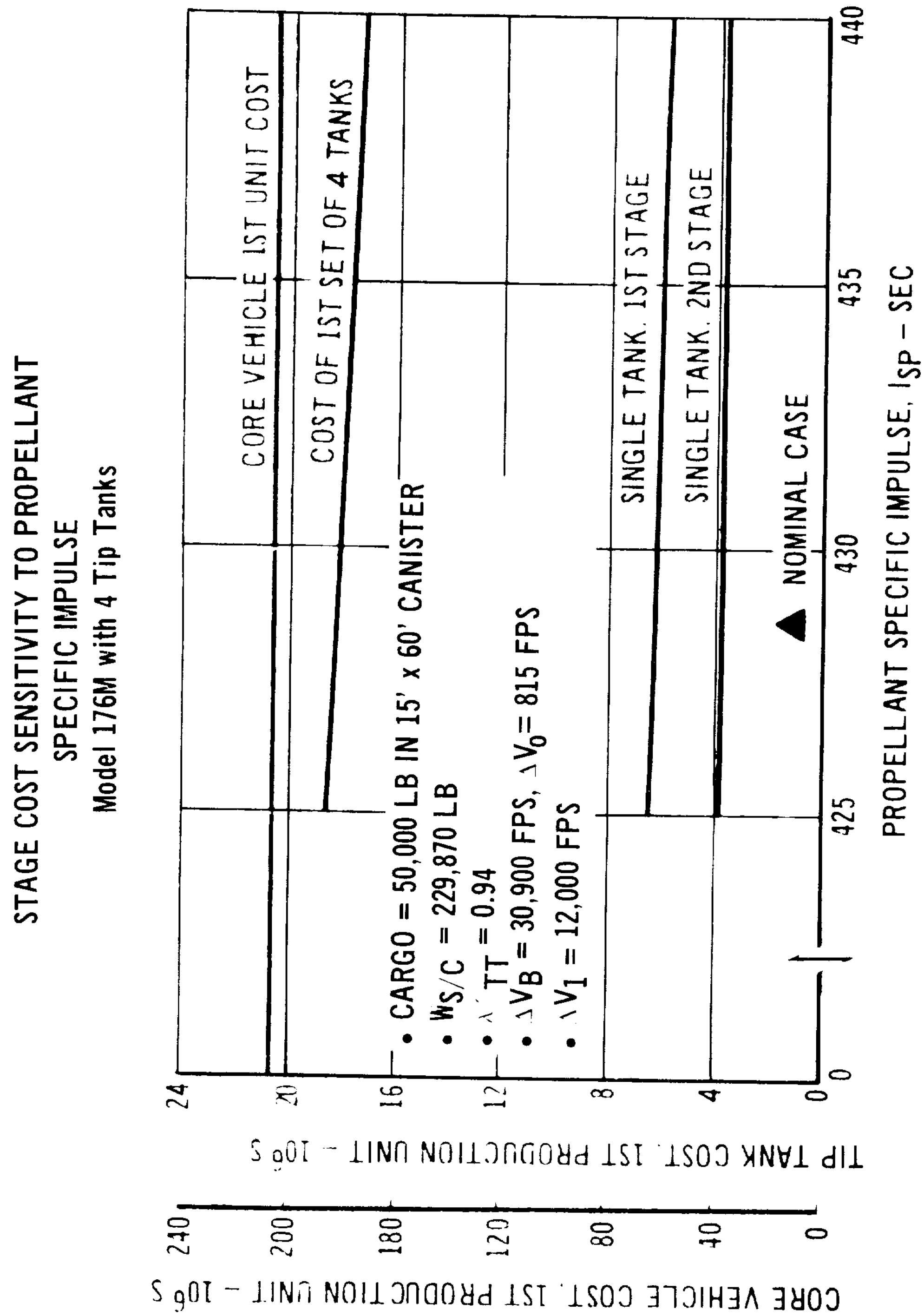

STAGE COST SENSITIVITY TO PROPELLANT SPECIFIC IMPULSE
Model 176M with 4 Tip Tanks
CORE VEHICLE 1ST UNIT COST
COST OF 1ST SET OF 4 TANKS
SINGLE TANK. 1ST STAGE
SINGLE TANK. 2ND STAGE
NOMINAL CASE
• CARGO = 50,000 LB IN 15' x 60' CANISTER
• $W_{S/C}$ = 229,870 LB
• λ'_{TT} = 0.94
• ΔV_B = 30,900 FPS, ΔV_0 = 815 FPS
• ΔV_1 = 12,000 FPS
PROPELLANT SPECIFIC IMPULSE, I_{SP} – SEC
0
425
430
435
440
TIP TANK COST, 1ST PRODUCTION UNIT – 10^6 \$
0
4
8
12
16
20
24
CORE VEHICLE COST, 1ST PRODUCTION UNIT – 10^6 \$
0
40
80
120
160
200
240

Figure 5–19

RECURRING COST SENSITIVITY TO PROPELLANT SPECIFIC IMPULSE

- MODEL 176M WITH 2 TIP TANKS

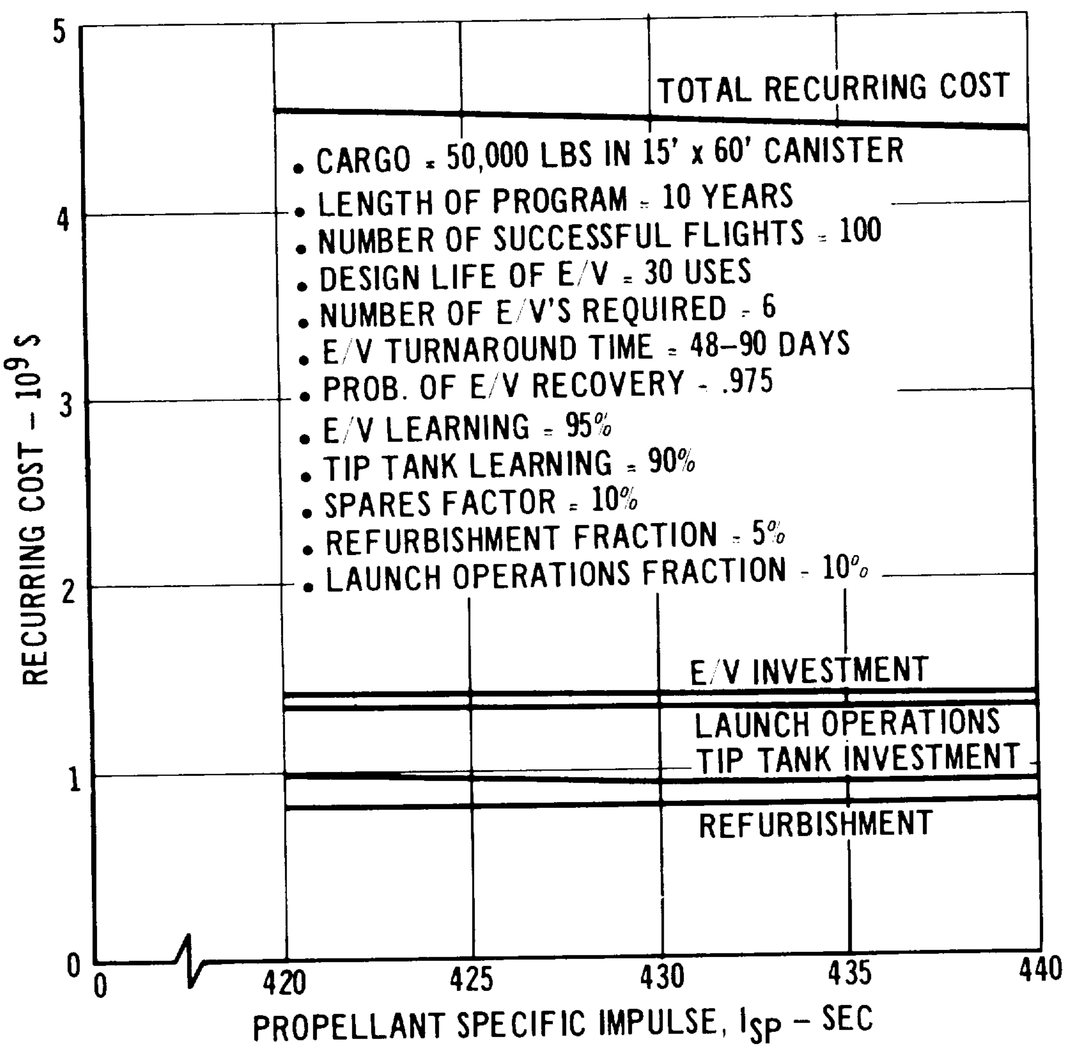


Figure 5-20

TOTAL RECURRING COST SENSITIVITY TO SPECIFIC IMPULSE

Model 176M With 4 Tip Tanks

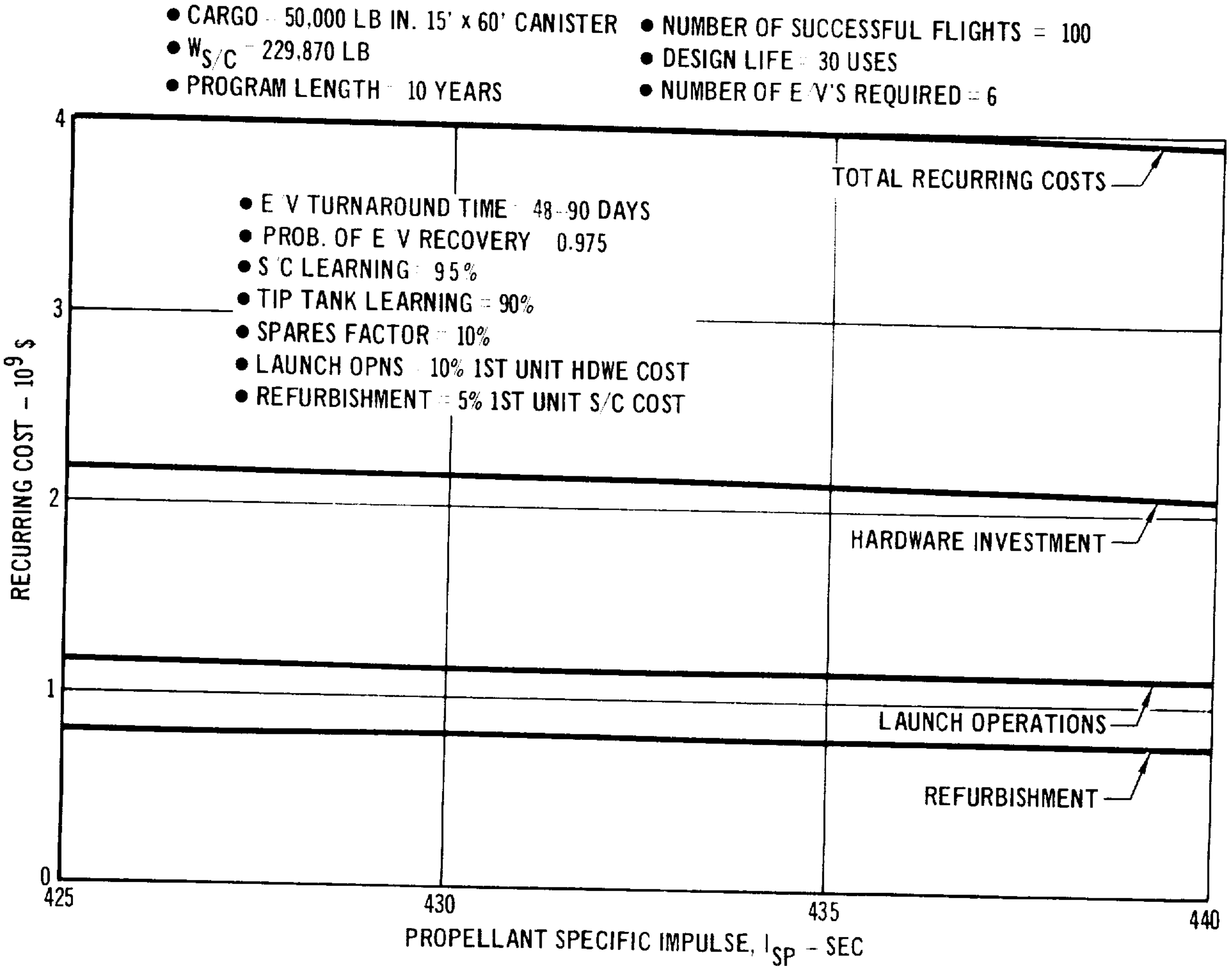


Figure 5-21